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J-2 ROCKET ENGINE DESIGN INFORMATION

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(NASA Contract NAS 8-19

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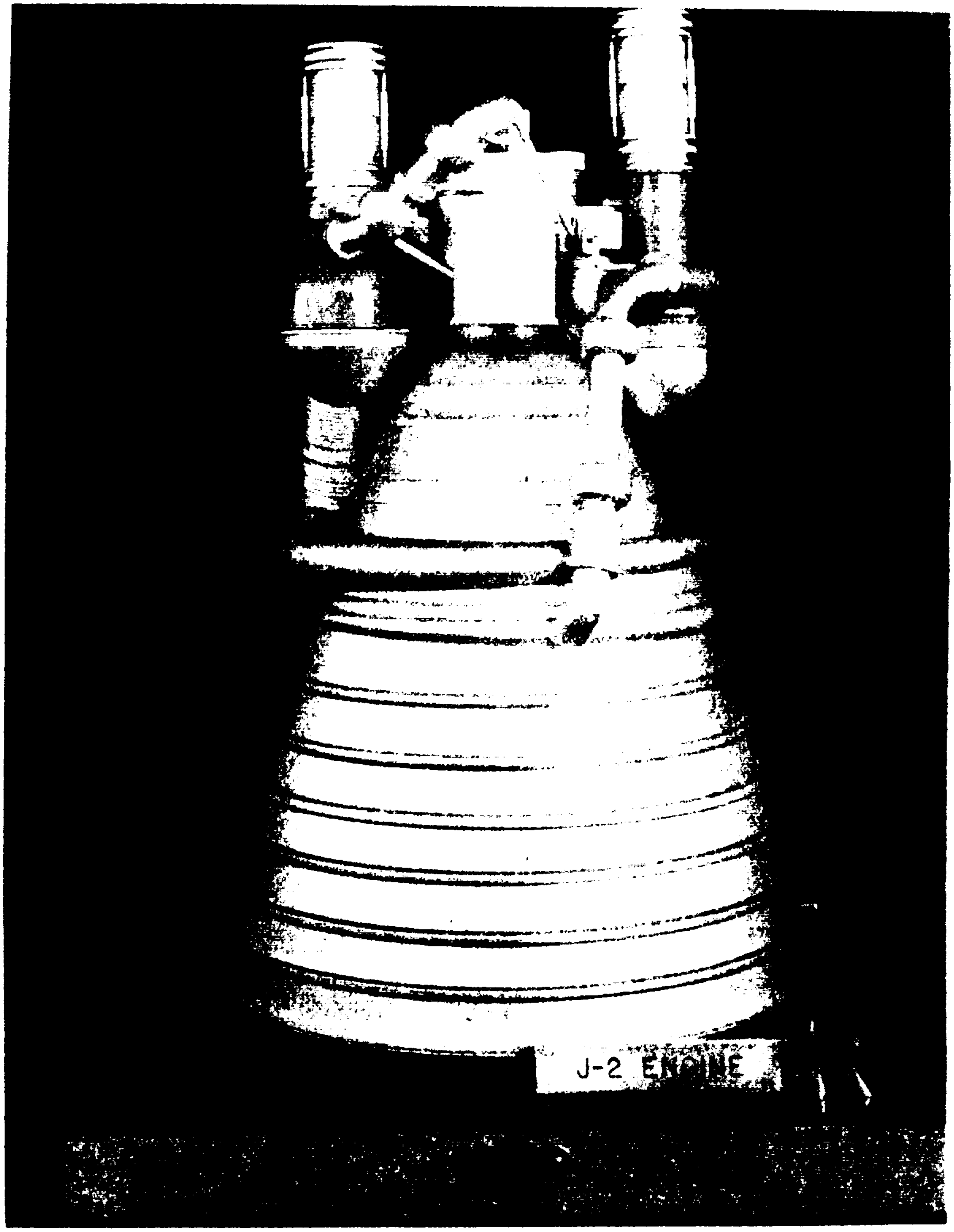

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Frontispiece

FOREWORD

Rocketdyne, a Division of North American Aviation, Inc., has prepared this report to present a summary of the characteristics of its model J-2 rocket engine being developed under the sponsorship of the National Aeronautics and Space Administration. This edition has been prepared especially for the information of the bidders on Saturn Stage II.

ABSTRACT

Initial design characteristics of the Rocketdyne model J-2, 200,000-lb-thrust oxygen-hydrogen rocket engine are presented. Engine operating characteristics have been estimated as completely as possible to provide a uniform basis for vehicle proposal effort.

[redacted] Abstract)

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ILLUSTRATIONS

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TABLES

INTRODUCTION

The use of rocket engines with high specific impulse offers advantages in both over-all system weight and performance. A requirement for a 200,000-lb-thrust oxygen-hydrogen engine has been established by the National Aeronautics and Space Administration for the second-stage propulsion of the Saturn vehicle.

In response to this requirement, Rocketdyne has applied its rocket engine and liquid hydrogen experience to the design of a large high-energy engine with high reliability as required for space missions. This activity resulted in a NASA-sponsored program to develop the engine, designated the J-2, through PFRT in February 1963 and Qualification in May 1964 (Fig. 1.1).

Although the J-2 engine has been designed primarily for operation at high altitude (vacuum), the engine can be operated at sea level for test or calibration in simple test stands which do not have altitude simulation equipment.

The following sections of this report present a summary of the design and operating characteristics of the J-2 rocket engine as they appear in the early phase of development. Engine operating characteristics, especially start and cutoff, are determined from a mathematical model of the propulsion system. These estimates are the most accurate now available, but must be understood to be preliminary and subject to change when engine test results become available.

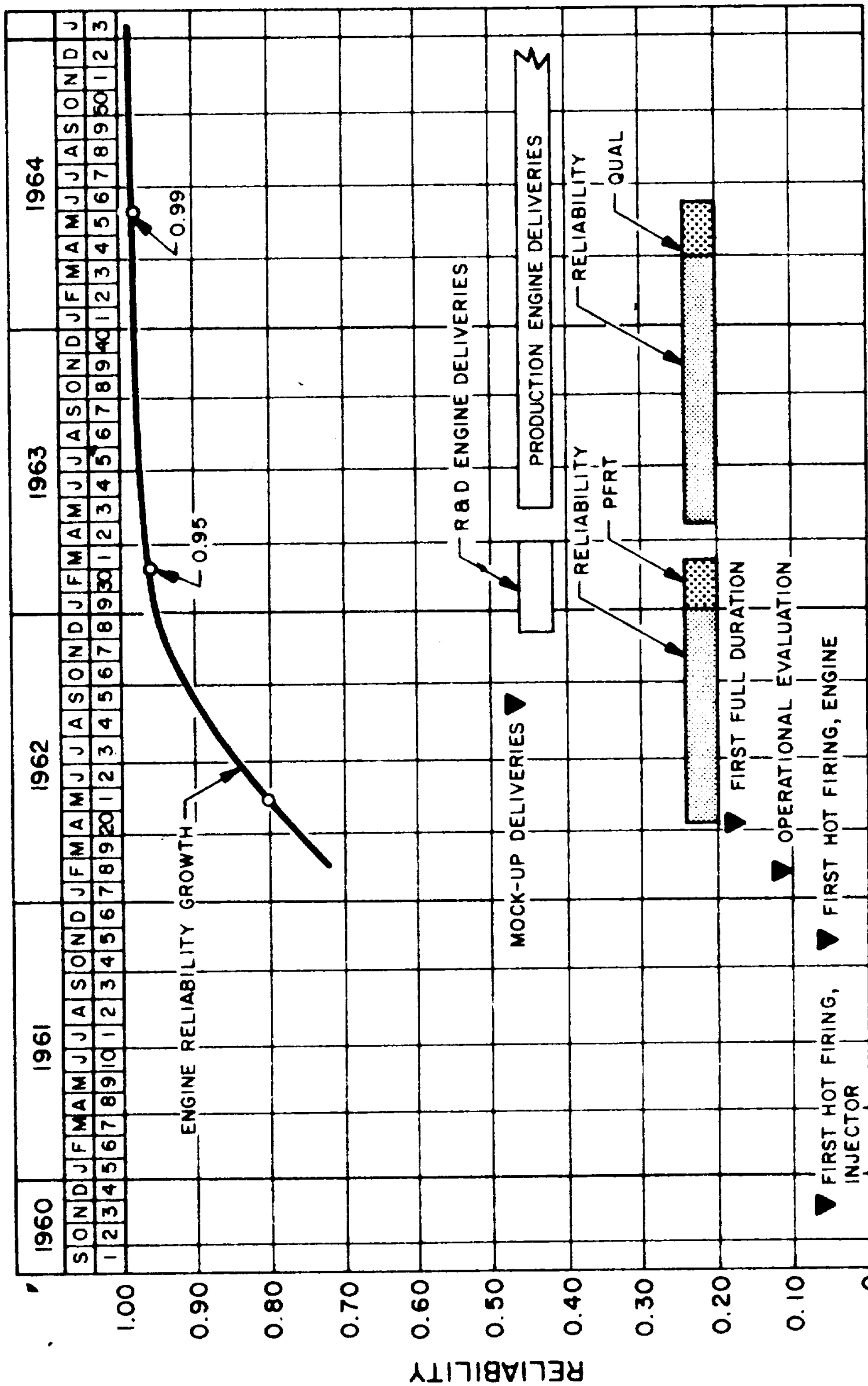

Figure 1.1. J-2 Engine Schedule

GENERAL DESCRIPTION

DESCRIPTION OF J-2 ROCKET ENGINE

The J-2 rocket engine is a 200,000-lb-thrust, high-energy, upper-stage propulsion system utilizing liquid oxygen and liquid hydrogen as propellants. The engine is designed to be used singly or in clusters.

Maximum performance is obtained in a configuration optimized from the standpoint of size and weight to be an engine with an envelope 80-in. in dia and 116-in. long (Fig. 2.1). A complete listing of performance characteristics is presented in Table 3.1. A listing of influence coefficients for computing performance under various conditions is presented in Table 3.2. A breakdown of engine weight is presented in Table 5.1.

The J-2 engine features a tubular-wall, bell-shaped thrust chamber, independently driven direct-drive turbopumps for liquid hydrogen and liquid oxygen, each operating at the most favorable speed, powered in series by a single gas generator utilizing the same propellants as the main thrust chamber. Engine contact with fluids is limited to the propellants and helium. There is a complete absence of lubricants or any other fluids which could freeze at low temperatures. A schematic is presented in Fig. 4.2.

An electrical control system sequences the engine start and shutdown. The sequencing operation is accomplished with solid-state logic elements because of their inherently high reliability.

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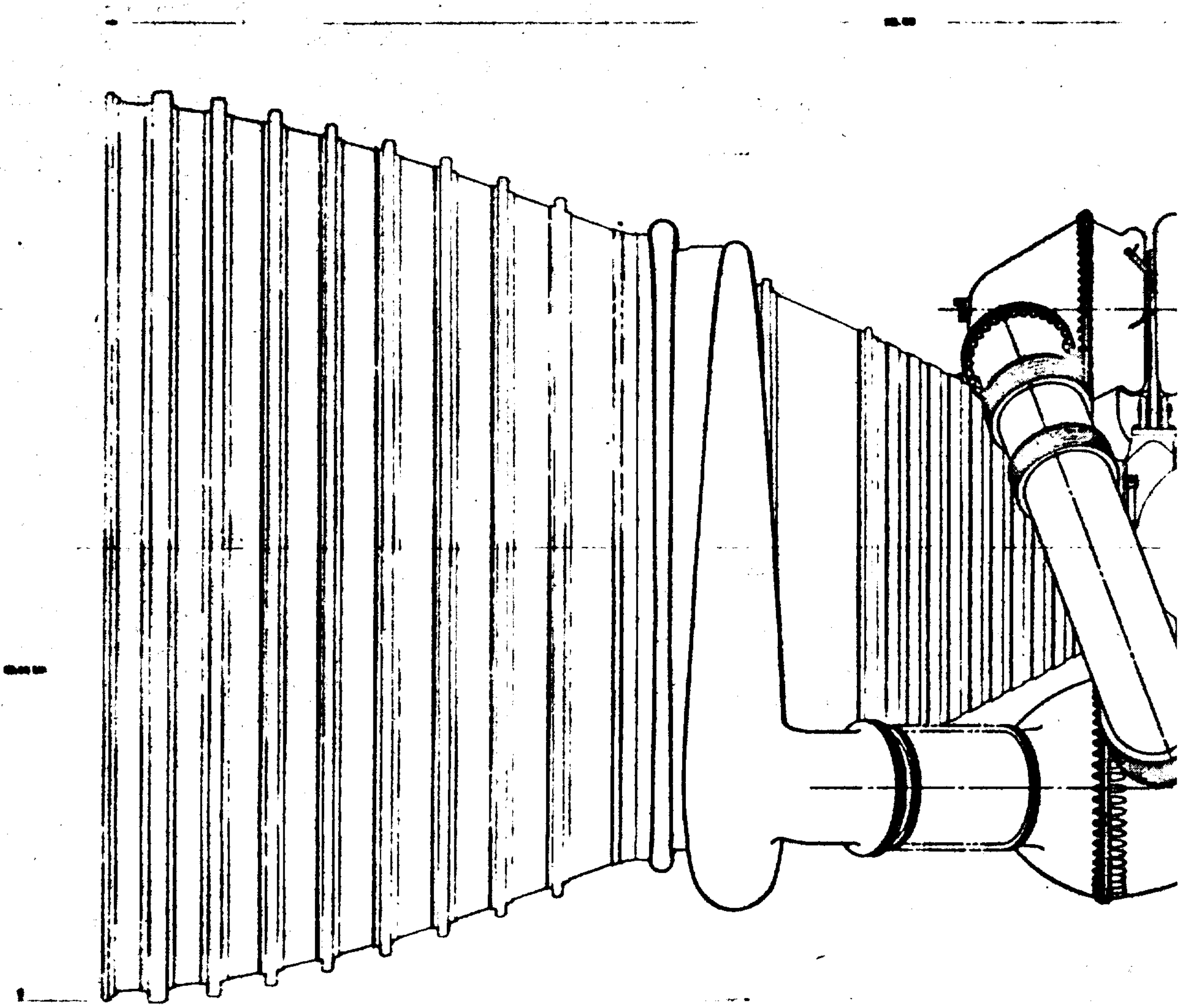

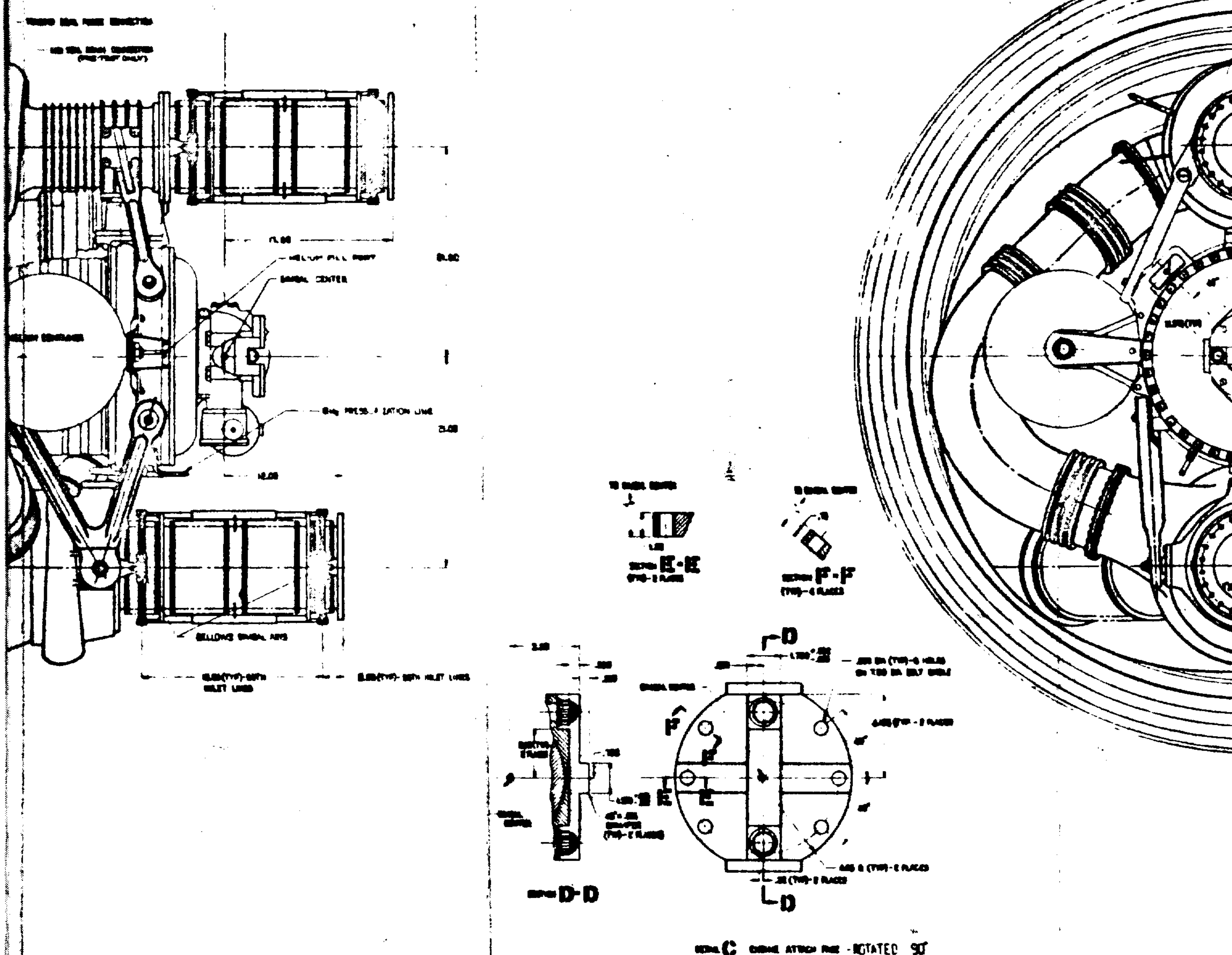

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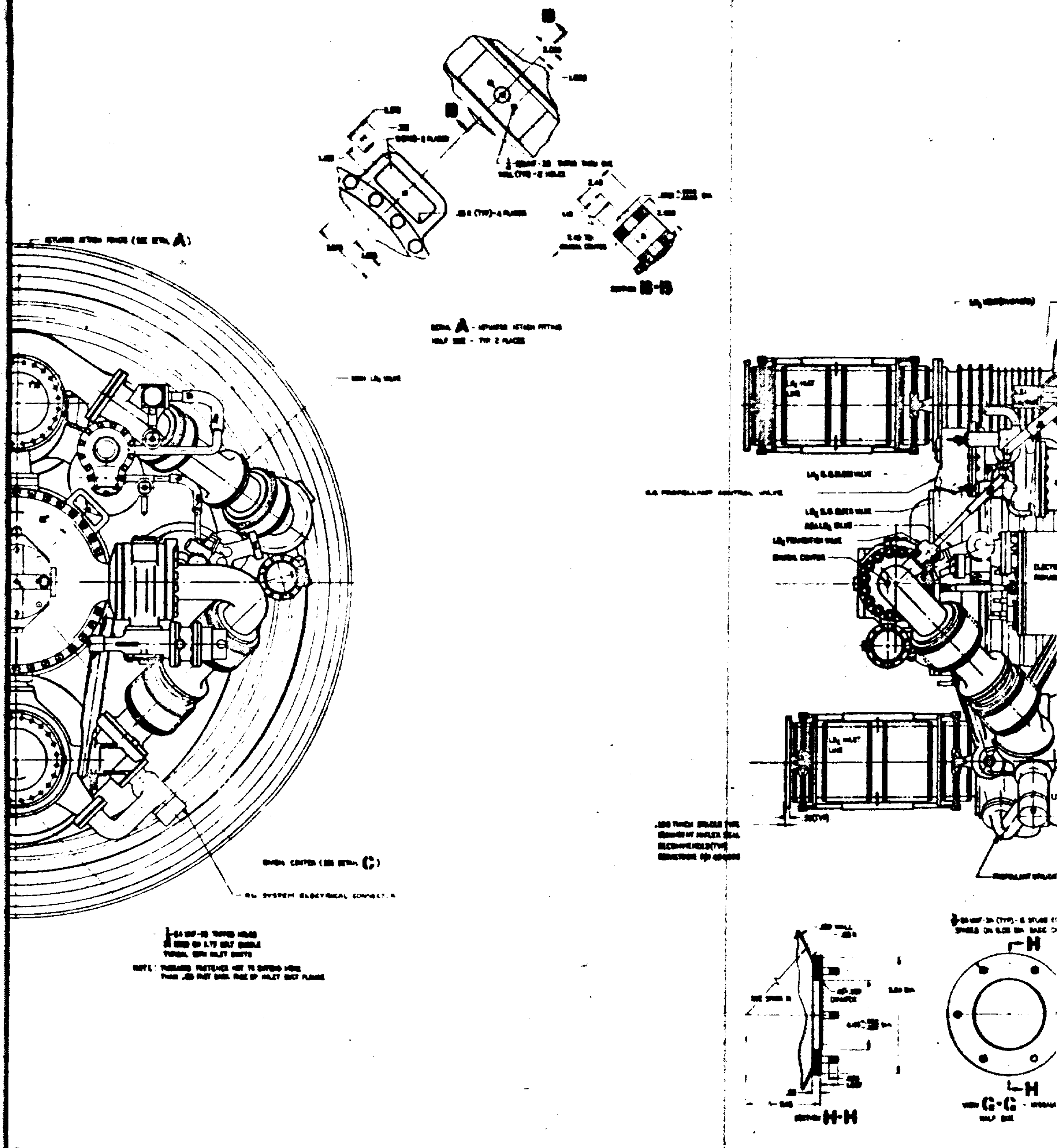

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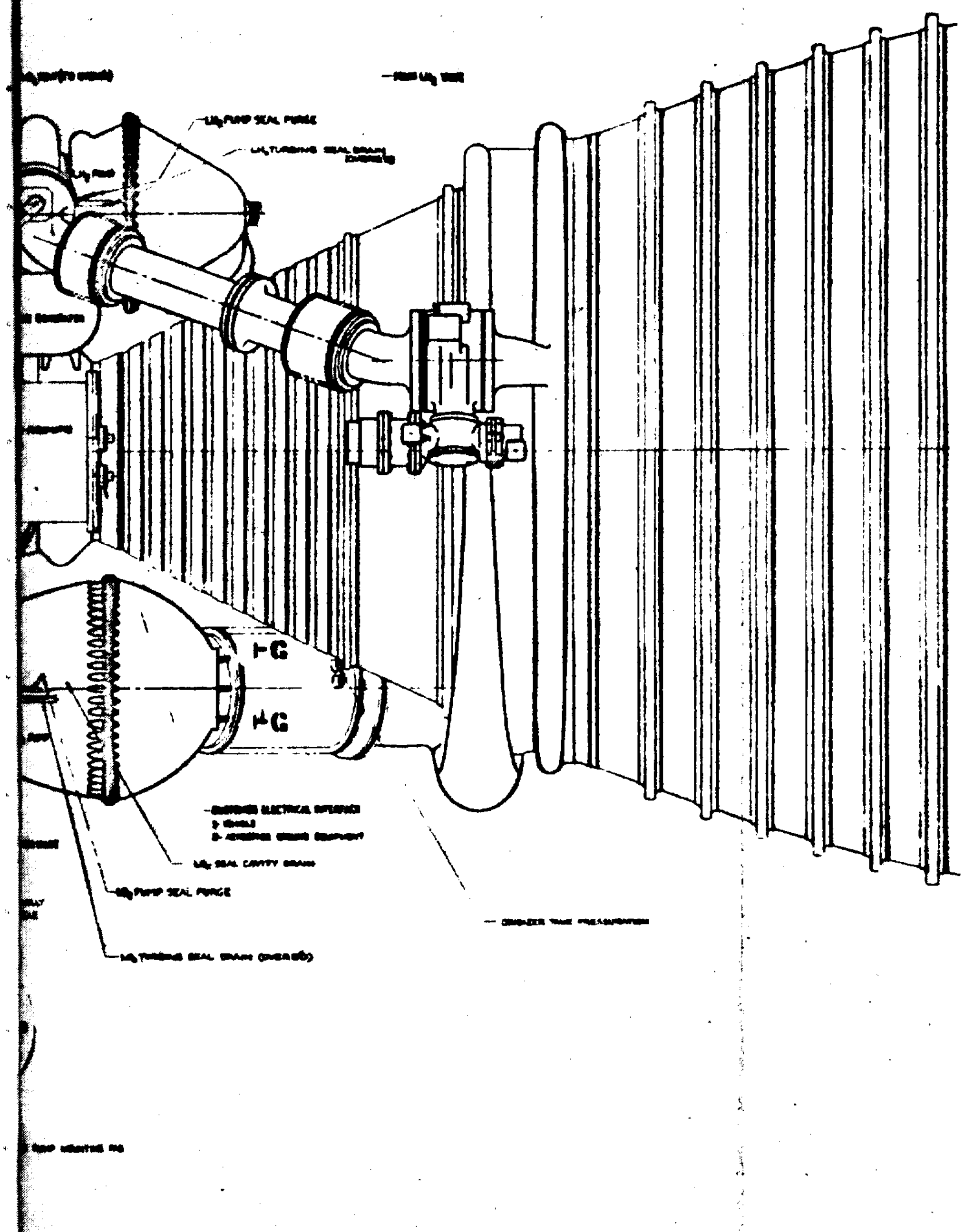

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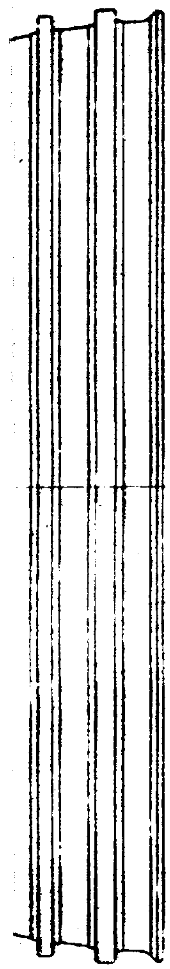

NOTE:
ALL DIMENSIONS NOMINAL AT +70° F

FOLDOUT FRAME 5

Figure 2.1. J-2 General Arrangement

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Flexible inlet bellows to the turbopumps are provided for attachment to rigid vehicle plumbing. Inlet bellows characteristics are given in section 7.

A high-speed, direct-drive, power takeoff is provided at the oxygen pump turbine. The pad characteristics are given in Fig. 10.1.

Flow measuring devices of the turbine meter type, for determining propellant flowrate, are provided in the high-pressure ducting downstream of each propellant pump.

A simplified propellant utilization system consists of a single, small, electrically operated valve bypassing propellant around the liquid oxygen pump. Characteristics of the propellant utilization valve servomotor are given in section 11.

Extensive use of welded and brazed joints has minimized the number of joints with seals, reducing possible leak points, and increasing system reliability. Dual seals with an intermediate bleed to a low-pressure region are used where seals are necessary.

The engine includes a gimbal bearing at the center of the injector dome for thrust vector control as described in section 6.

The Frontispience and Fig. 2.2, 2.3, and 2.4 are general views of the J-2 engine.

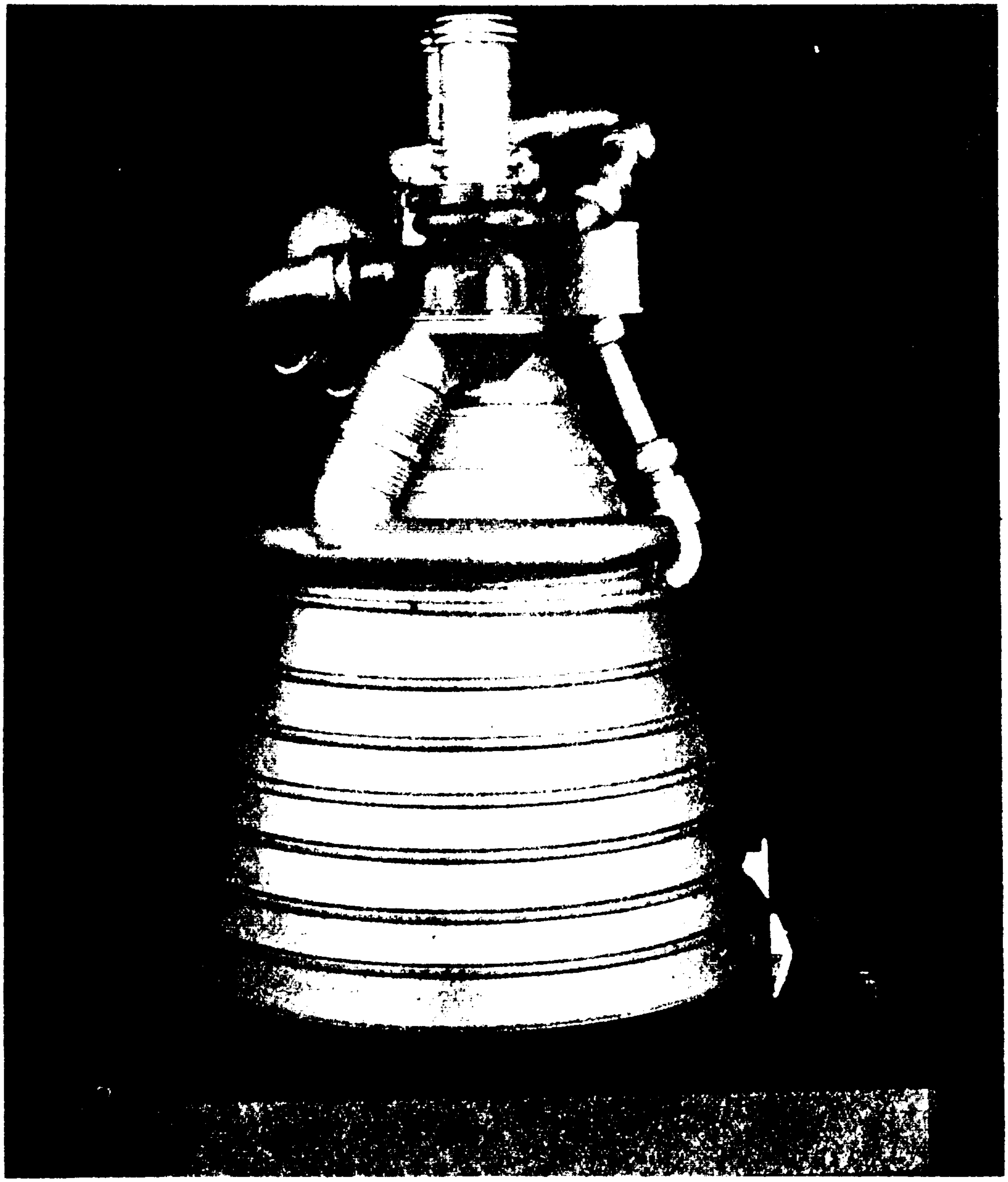

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Figure 2.2. J-2 Engine, View 1

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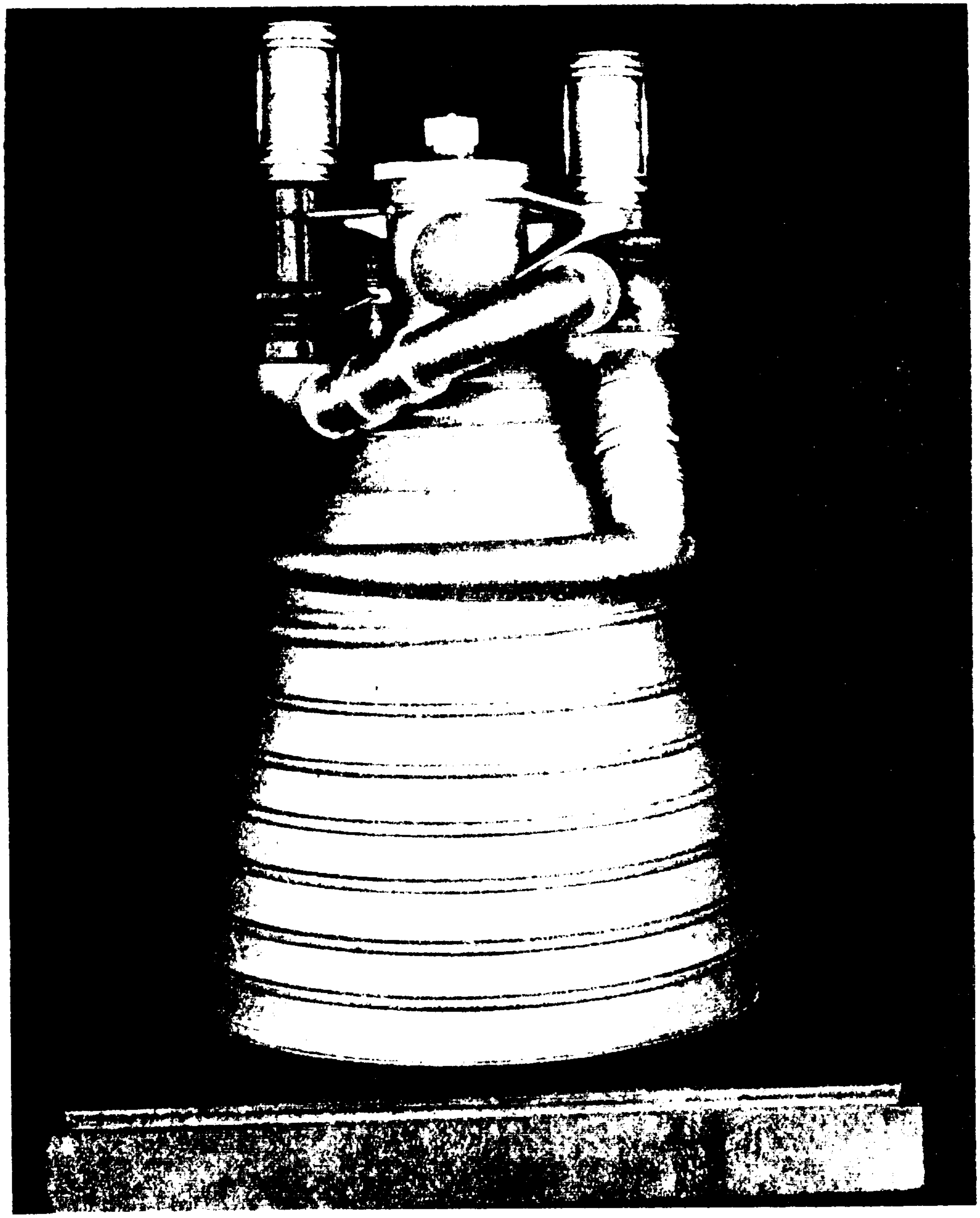

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Figure 2.3. J-2 Engine, View 2

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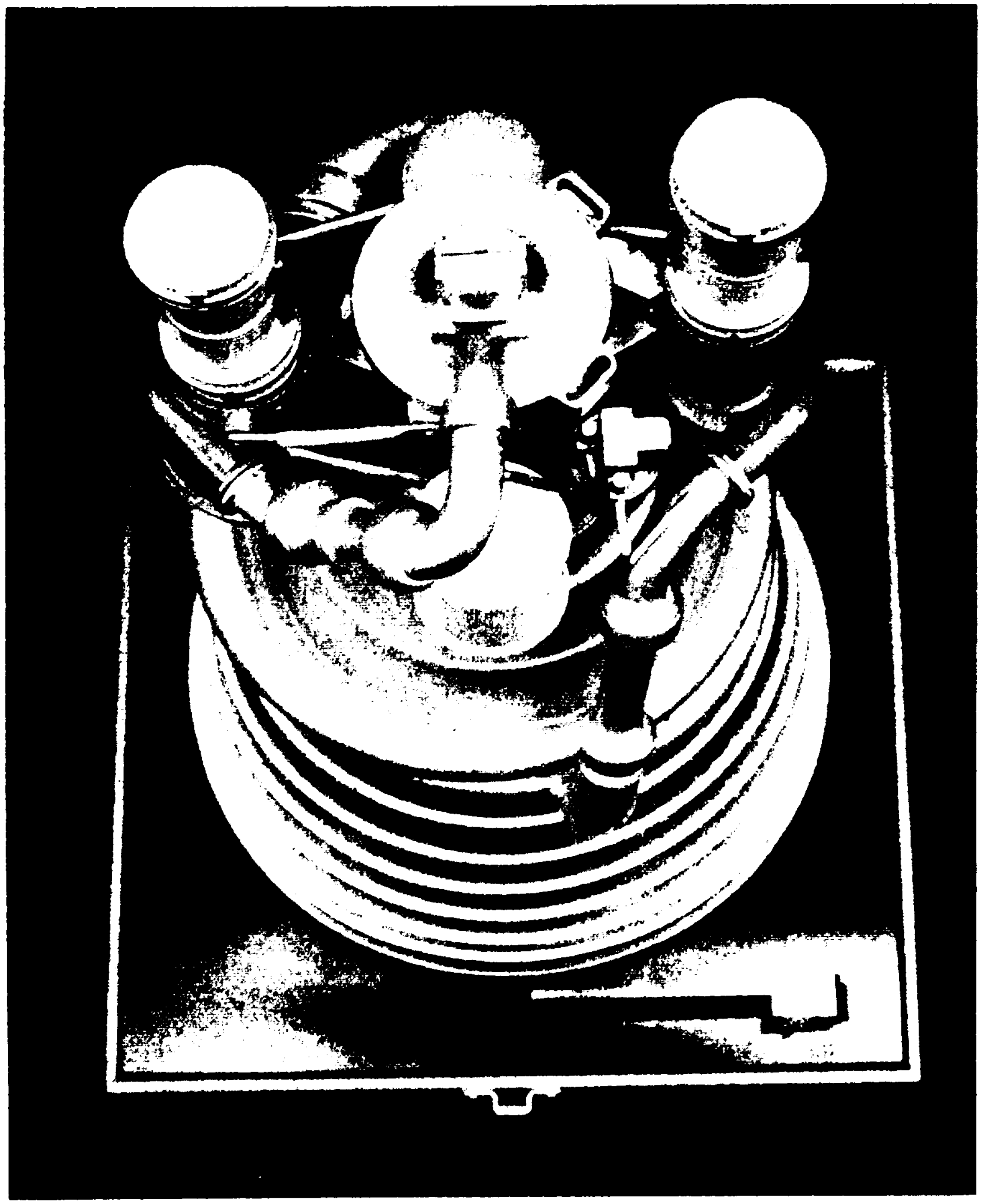

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Figure 2.4. J-2 Engine, View 3

PERFORMANCE

PERFORMANCE PARAMETERS

The nominal performance parameters for the J-2 engine are presented in Table 3.1. The values are for nominal conditions during mainstage only.

Curves are presented in Fig. 3.1 and 3.2 defining the variations in specific impulse and thrust with altitude. These curves should be used for approximation purposes only.

As the propellant utilization valve is moved from the open to the closed position, a variation in engine thrust will result. The data to calculate this effect are included in Table 3.2. Figure 3.3 and 3.4 can be used for rough approximations.

The engine balance presented in Table 3.1 is based upon an auxiliary power drive of 30 bhp, assuming a hydraulic pump is mounted to power a hydraulic gimbal actuation system. A discussion of the pump drive pad is presented in section 10.

Oxygen gas for vehicle tank pressurization is provided by an oxygen heat exchanger. Hydrogen gas for pressurizing the vehicle hydrogen tank is provided from a thrust chamber jacket bleed. Changes in tank pressurization flowrate will affect engine balance (section 10).

Engine specific impulse and mixture ratio do not include the propellants diverted for tank pressurization.

TABLE 3.1

J-2 ENGINE NOMINAL PERFORMANCE PARAMETERS
(VALUES AT MAINSTAGE ONLY)

Propellants	
Liquid Oxygen Density*, lb/cu ft	71.38
Liquid Hydrogen Density*, lb/cu ft	4.42
Thrust (altitude), lb	200,000
Specific Impulse (altitude), sec	426
Mixture Ratio, o/f	5.00
Rated Duration, sec	250
Oxidizer Weight Flowrate (pump inlet), lb/sec	395.15
Fuel Weight Flowrate (pump inlet), lb/sec	81.24
Chamber Pressure (nozzle stagnation), psia	632
Expansion Area Ratio	27.5
Oxidizer Pump (centrifugal)	
Inlet Pressure (nominal), psia	32
Discharge Pressure, psia	1019
Developed Head, ft	1992
Volumetric Flowrate, gpm	2893
Weight Flowrate, lb/sec (includes nominal propellant utilization bypass)	460.15
Minimum NPSH (pump inlet), ft	25 (12.5 psi)
Horsepower, bhp	2604
Nominal Speed, rpm	8947

*Saturated liquid at normal boiling point

TABLE 3.1

(Continued)

Fuel Pump (axial)	
Inlet Pressure (nominal), psia	25
Discharge Pressure, psia	1088
Developed Head, ft	34,615
Volumetric Flowrate, gpm	8250
Weight Flowrate, lb/sec	81.24
Minimum NPSH (pump inlet), ft	130 (4 psi)
Horsepower, bhp	6243
Speed, rpm	26,455
Oxidizer Turbine	
Inlet Pressure, psia	98.8
Outlet Pressure, psia	28.4
Inlet Temperature, F	839
Outlet Temperature, F	688
Auxiliary Power Available, bhp	30
Fuel Turbine	
Inlet Pressure, psia	627.7
Outlet Pressure, psia	99.2
Inlet Temperature, F	1200
Outlet Temperature, F	839
Gas Generator	
Chamber Pressure (injector end), psia	634.5
Oxidizer Weight Flowrate, lb/sec	3.15
Fuel Weight Flowrate, lb/sec	5.33

TABLE 3.1

(Continued)

Hydrogen Tank Pressurization	
Weight Flowrate, lb/sec	3.00
Temperature, F	-260
Oxygen Tank Pressurization	
Weight Flowrate, lb/sec	3.90
Temperature, F	-170

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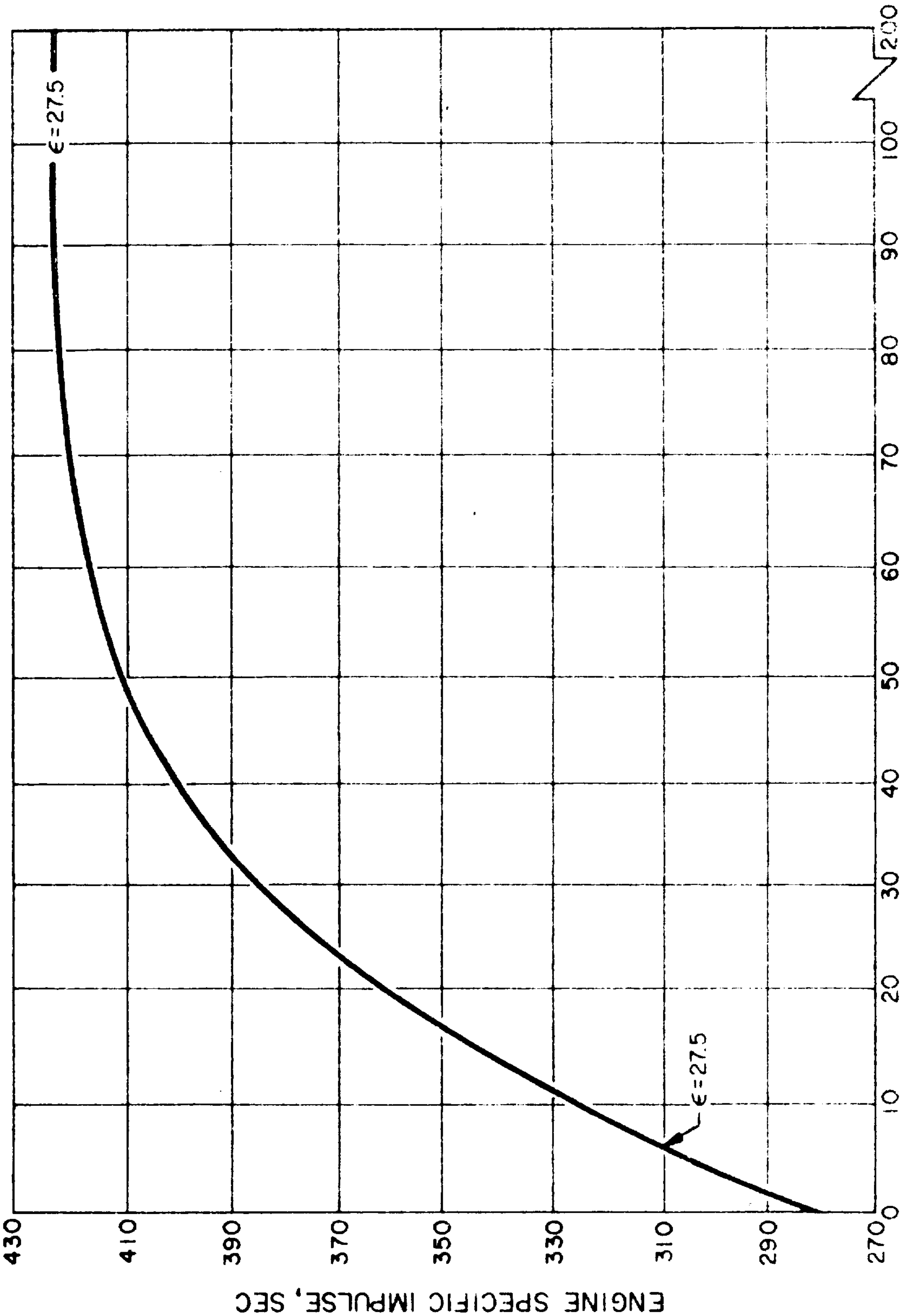

Figure 3.1 . Specific Impulse vs Altitude for J-2 Engine

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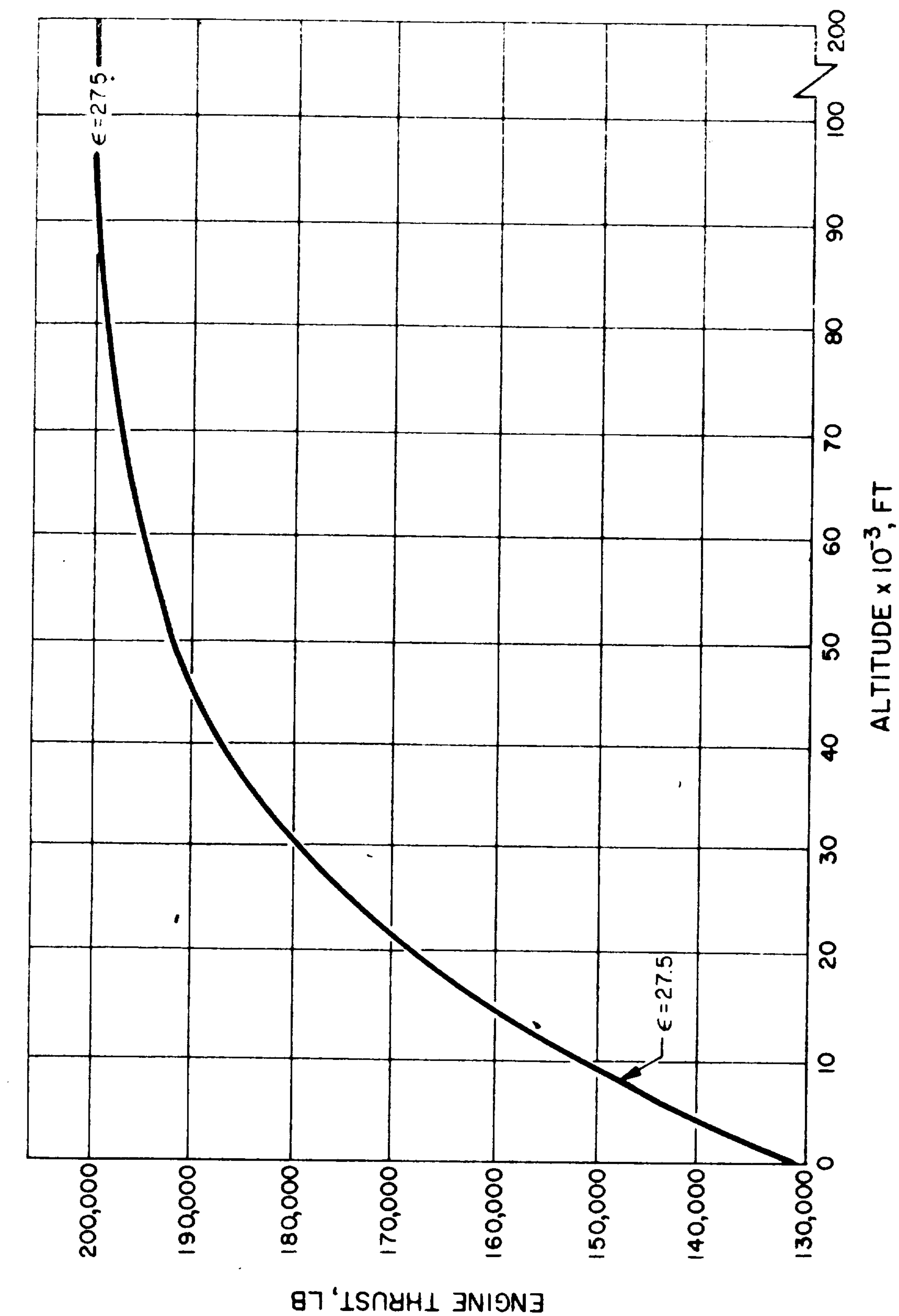

Figure 3.2. Thrust vs Altitude for J-2 Engine

TABLE 3.2

J-2 ENGINE INFLUENCE COEFFICIENTS[1]

Dependent Variables and Nominal Values	Independent Variables and Nominal Values					
A +1 percent change of → causes the following percent change in ↓	Fuel Density, 4.42 lb/cu ft	Oxidizer Density 71.38 lb/cu ft	Fuel Pump Inlet Pressure, 25.0 psia	Oxidizer Pump Inlet Pressure 32.0 psia	Propellant Utilization Control Setting	C* Correction[2]
Thrust, 200,000 lb	+0.2402	+0.7502	+0.0041	+0.0280	+0.0486	+1.0744
Specific Impulse, 426.0 sec	+0.0942	-0.0841	+0.0021	-0.0033	-0.0057	+1.0050
Fuel Flowrate, 78.24 lb/sec	+0.6832	+0.2969	+0.0142	+0.0105	+0.0182	+0.0834
Oxidizer Flowrate, 391.25 lb/sec	+0.0386	+0.9418	-0.0005	+0.0354	+0.0615	+0.0666
Mixture Ratio, 5.00 o/f	-0.6446	+0.6448	-0.0147	+0.0250	+0.0433	-0.0168

[1] Engine influence coefficients result from a linearized solution of a set of steady-state equations which describe the operation of an engine. Each influence coefficient is expressed in percentage form and represents the effect upon an engine dependent variable of a +1 percent change in an engine independent variable. A coefficient preceded by a positive (+) sign (or no sign) indicates that an increase in the independent variable results in an increase in the dependent variable; a coefficient preceded by a negative (-) sign indicates that an increase in the independent variable results in a decrease in the dependent variable.

[2] This is optional for increased accuracy, and is to be used with the other independent variables to compute changes in the dependent variables due to C* nonlinearity in the system.(Page 3.14)

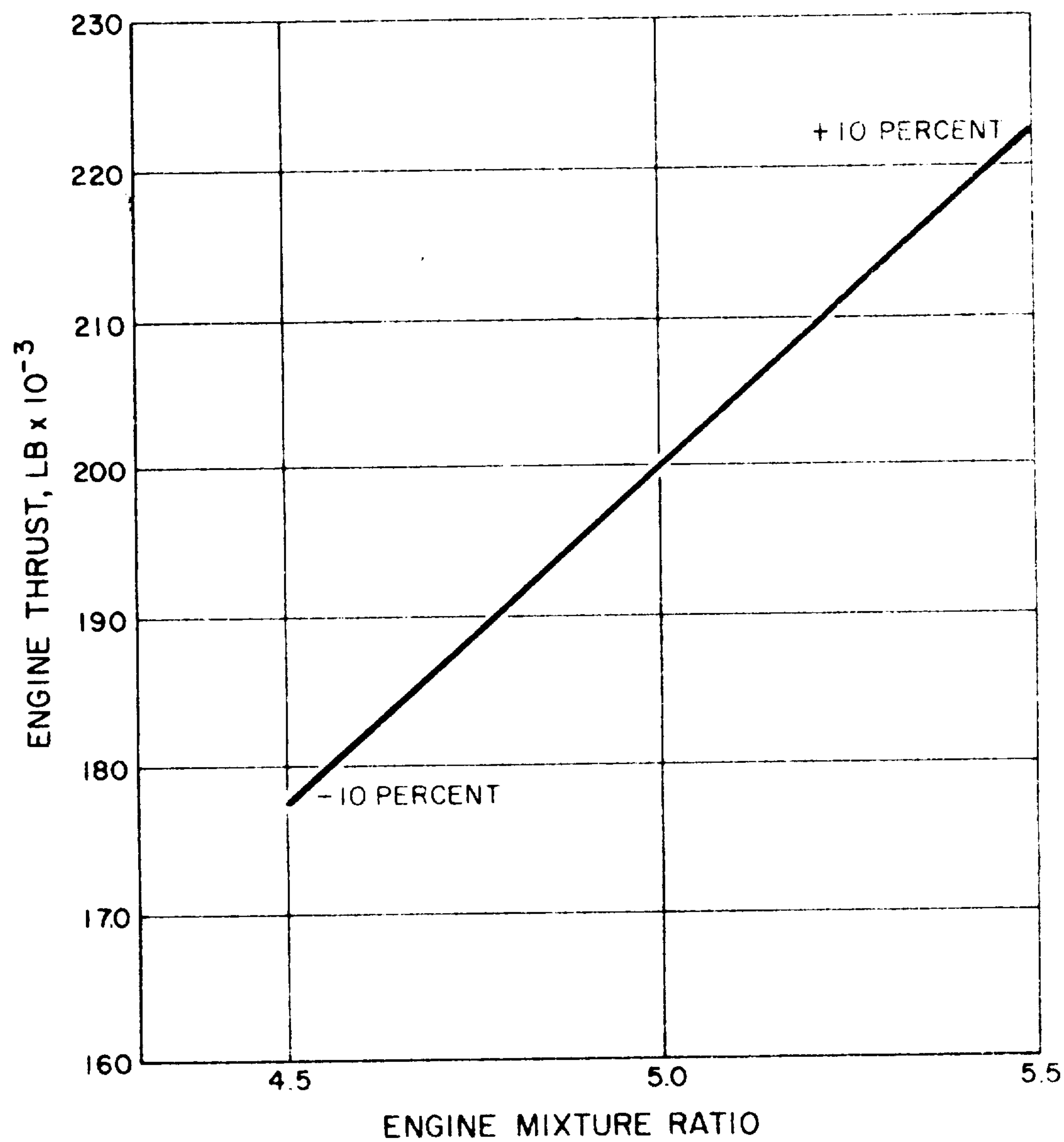

Figure 3.3. Engine Thrust vs Engine Mixture Ratio for Variation in Propellant Utilization Valve Position

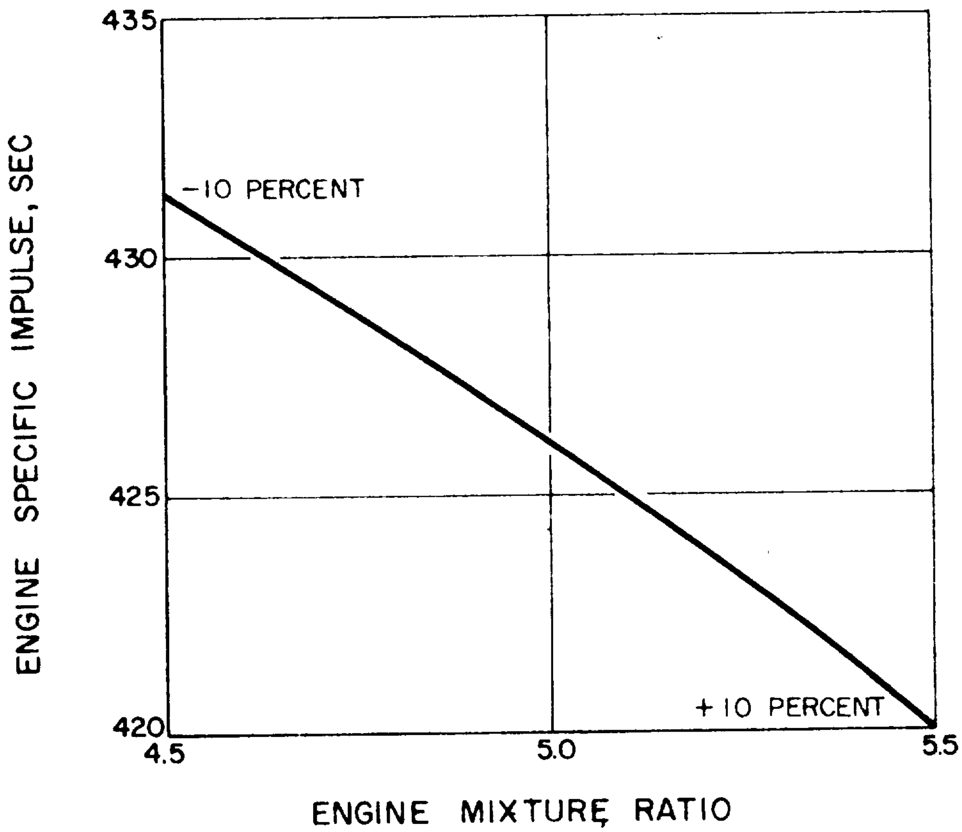

Figure 3.4. Engine Specific Impulse vs Engine Mixture Ratio for Variation in Propellant Utilization Valve Position

J-2 ENGINE PERFORMANCE AT VARIOUS ALTITUDES

The J-2 engine is designed for upper-stage vheicle application, and is optimized for vacuum operation with a relatively large nozzle expansion area ratio of 27.5:1. However, this engine has the advantage of capability of operation at altitudes down to sea level without the occurrence of jet separation in the nozzle. This was achieved by advanced nozzle design. The J-2 engine can be test fired, acceptance tested, and fired in clusters at sea level on ordinary test stands which do not have altitude-simulating aspirators. This will result in greater accuracy in test measurements, gimbal system checkout, and considerably greater convenience. The J-2 engine performance at various altitudes is indicated in Fig. 3.1 and 3.2.

THRUST VARIATION RESULTING FROM PROPELLANT UTILIZATION VALVE OPERATION

The J-2 engine design provides for mixture ratio variations of ± 0.5 units of mixture ratio. This variation produces a power variation to the turbopump resulting in a thrust variance. This variation with respect to engine mixture ratio is presented in Fig. 3.3. For detailed analysis, the influence coefficients will yield more accurate parameters. The mixture ratio limits under any condition are not less than 4.0 or more than 6.0.

INFLUENCE COEFFICIENTS

Influence coefficients are presented in Table 3.2 These influence coefficients possess adequate accuracy over the entire design operating range of the J-2 engine for application to preliminary vehicle analyses.

SAMPLE CALCULATION USING TABLE 3.2 INFLUENCE COEFFICIENTS

The use of Table 3.2 influence coefficients for determining thrust of the engine where operating conditions are

Fuel density	4.34 lb/ft^3
Oxidizer density	71.38 lb/ft^3
Fuel pump inlet pressure	26 psia
Oxidizer pump inlet pressure	37 psia
Propellant utilization	setting required for +8.0 percent mixture ratio shift

is shown in the following sample calculation.

Because the influence coefficients are linear, the total effect of several influences acting simultaneously on an engine can be determined by adding the individual effects of each influence. The change in engine thrust would be

$$\frac{F_E - F_{E_N}}{F_{E_N}} = \frac{\rho_E - \rho_{F_N}}{\rho_{F_N}} \; F_{\rho_E} \; \frac{\rho_o - \rho_{o_N}}{\rho_{o_N}} \; F_{\rho_o} + \frac{P_F - P_{F_N}}{P_{F_N}} \; F_{P_F} + \frac{P_o - P_{o_N}}{P_{o_N}} \; F_{P_o} + \frac{PU - PU_N}{PU_N} \; F_{PU} \qquad (1)$$

where F_E, ρ_F, ρ_o, P_F, P_o, and PU are the actual values of engine thrust, fuel density, oxidizer density, fuel pump inlet pressure, oxidizer pump inlet pressure and propellant utilization control setting respectively.

F_{E_N}, ρ_{F_N}, ρ_{E_N}, P_{F_N}, P_{o_N}, and PU_N are the nominal values of these parameters, as listed in the table of influence coefficients.

F_{ρ_F}, F_{ρ_o}, F_{P_F}, F_{P_o}, and F_{PU} are the influence coefficients for engine thrust found in the appropriate columns of the table of influence coefficients.

The calculation for the example stated above would be as follows:

The percentage change in propellant utilization (PU) control setting to give a specified change in mixture ratio is found from the equation

$$\frac{MR_E - MR_{E_N}}{MR_{E_N}} = \frac{PU - PU_N}{PU_N} \; (MR_{PU}) \qquad (2)$$

where MR_{PU} is the influence coefficient for engine mixture ratio found in the PU control setting column. For a +8.0 percent change in

$$\frac{MR_E - MR_{E_N}}{MR_{E_N}} \; , \; \frac{PU - PU_N}{PU_N} = \frac{+8.0}{MR_{PU}} = \frac{+8.0}{0.0433} = 184.8 \text{ percent} = +1.848$$

Therefore, substituting appropriate values in Eq. 1 gives:

$$\frac{F_E - 200{,}000}{200{,}000} = \frac{4.34 - 4.420}{4.420}(+0.2402) + \frac{71.38 - 71.38}{71.38}(+0.7502) +$$

$$\frac{26.00 - 25.00}{25.00}(+0.0041) + \frac{37.00 - 32.00}{32.00}(+0.0280) +$$

$$1.848\ (+0.0486)$$

$$\frac{F_E - 200{,}000}{200{,}000} = -(0.01810)\ (+0.2402) + (0)\ (+0.7502) +$$

$$(0.0400)\ (+0.0041) + (0.1563)\ (+0.0280) +$$

$$(1.848)\ (+0.0486) = +\ 0.09001 \text{ or } 9.001 \text{ percent}$$

Therefore

$$F_E = (200{,}000)\ (+0.09001) + 200{,}000$$

$$F_E = +18{,}002 + 200{,}000 = +\ 218{,}002 \text{ lb}$$

The incremental thrust change has been found to be +18,002 lb for the conditions stated, yielding a final engine thrust of 218,002 lb.

Nonlinear Corrections

A special computational procedure has been devised to extend the usefulness of engine influence coefficients. This technique is used to allow non-linear corrections to be made for certain parameters where the linear approximation is not sufficiently accurate. An example of this method is the c* correction. In this case, a plot of c* correction vs the change in engine mixture ratio is included with the table of influence coefficients, Fig. 3.5.

The change in engine mixture ratio is computed for the changes in propellant densities, pump inlet pressures, and with the assumption that the c* correction is zero. With this change in engine mixture ratio, the c* correction is read from the curve. This value of c* correction is used with the other independent variables to compute the changes in the remaining dependent variables.

For example, the change in engine mixture ratio used to effect the 9.001-percent thrust change in the preceding example was 8.00 percent, the c* correction from Fig. 3.5 is -0.05 percent. The true change in engine thrust is therefore:

$$\begin{aligned}(\text{percent change in } F_E) &= +9.001 - 0.05\ (+1.0744) \\ &= +9.001 - 0.054 = +8.947 \text{ percent}\end{aligned}$$

Similarly, other nonlinear corrections would be applied as additional terms in the summation of effects in an iterative procedure.

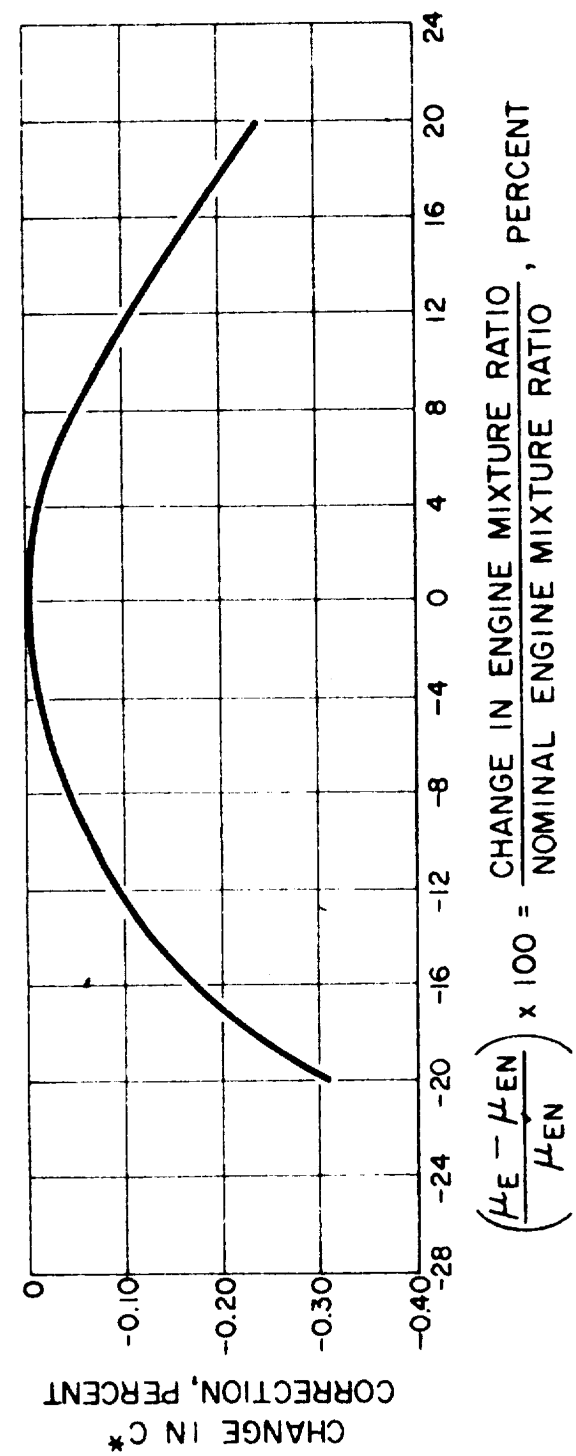

Figure 3.5. Characteristic Velocity Correction Curve

OPERATING CHARACTERISTICS

OPERATING CHARACTERISTICS

OPERATIONAL SEQUENCE

The operational sequencing of the engine is as follows, and is presented in Fig. 4.1. An engine schematic is presented in Fig. 4.2.

Start Sequence

The tank-head start technique, wherein the available tank pressures are utilized to establish gas generator operation and subsequent mainstage performance, is used to start the engine system. This start sequence is described in detail in the following paragraphs.

The vehicle programmer furnishes a start signal which energizes the spark exciters to initiate sparks in the gas generator and in the thrust chamber augmented spark ignition (ASI) system. At the same time, the helium supply solenoid is energized, opening the regulator and supplying gas to the system. The entire helium supply flows through a check valve, located in the pneumatic package. (This check valve is provided to allow the propellant valves to remain open during engine operation in event of gas supply system failure.) Purge gas flows into the thrust chamber oxidizer dome, the gas generator bleed valve closes, control gas flows to the closing control ports of the main oxygen valve, the gas generator oxygen transition valve, the gas generator valve, and the main hydrogen valve. The control solenoid that supplies gas to the ASI oxygen and main

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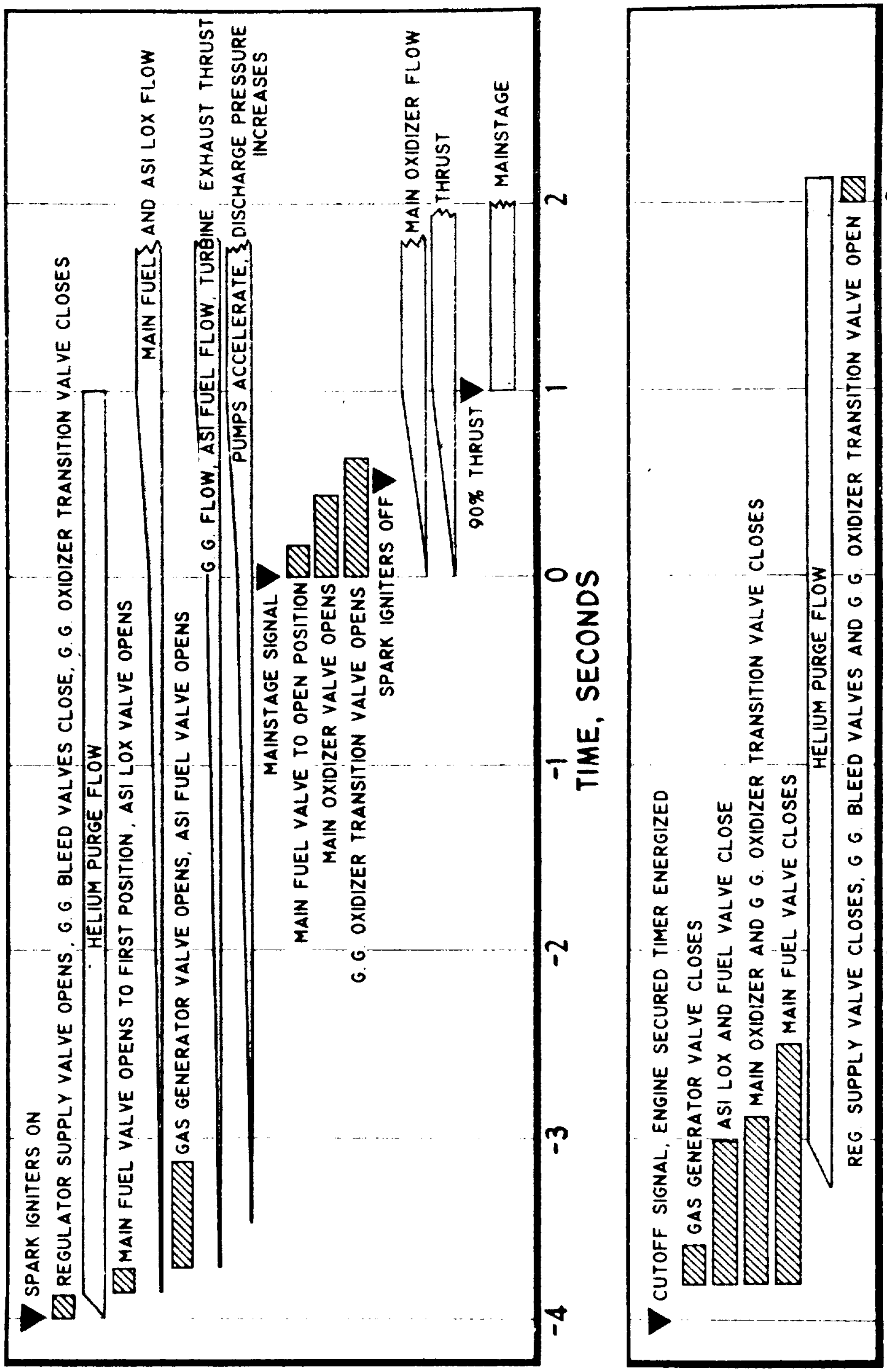

Figure 4.1. J-2 Start and Cutoff Sequence

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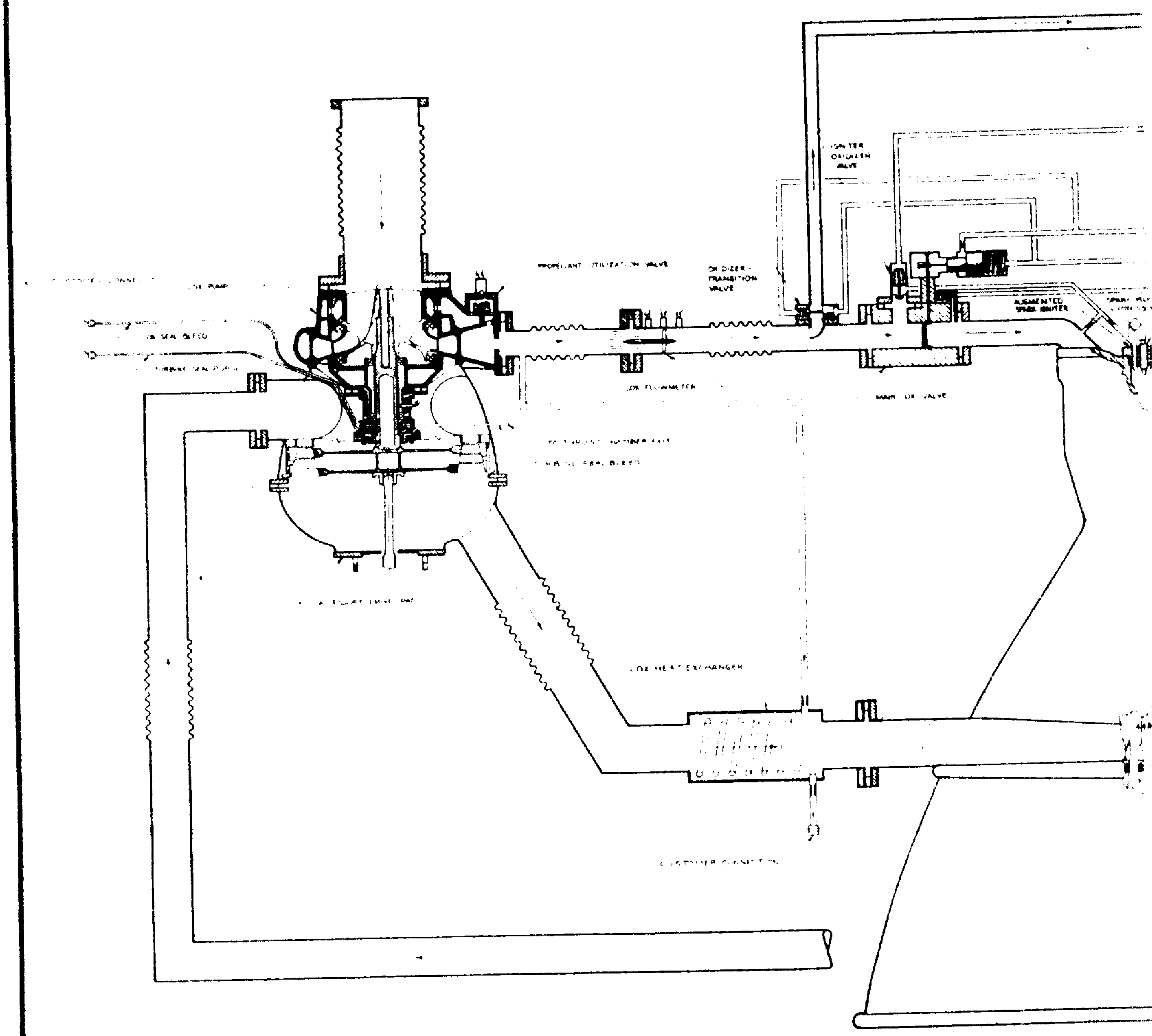
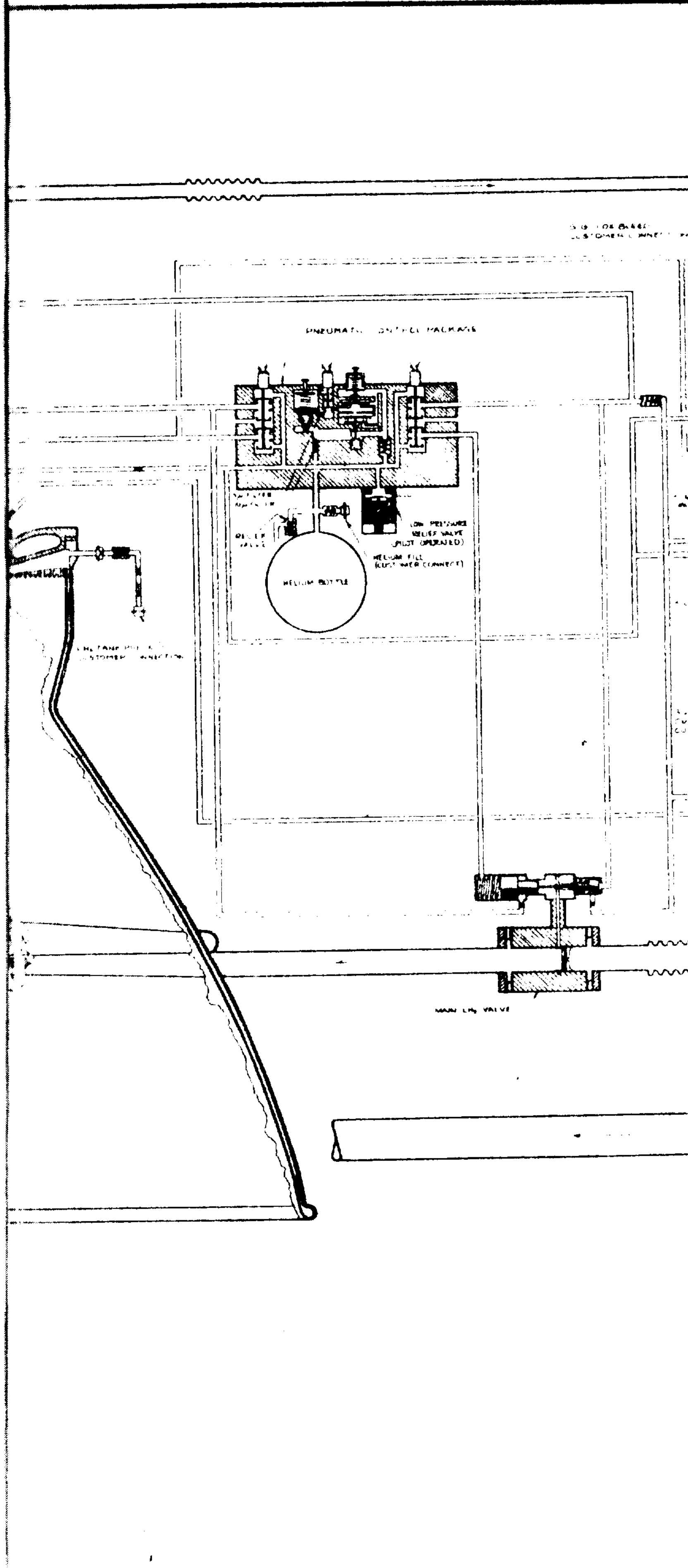
[redacted]



PROPELLANT UTILIZATION VALVE
OXIDIZER TRANSITION VALVE
IGNITER OXIDIZER VALVE
AUGMENTED SPARK IGNITER
LOX FLOWMETER
MAIN LOX VALVE
LOX HEAT EXCHANGER



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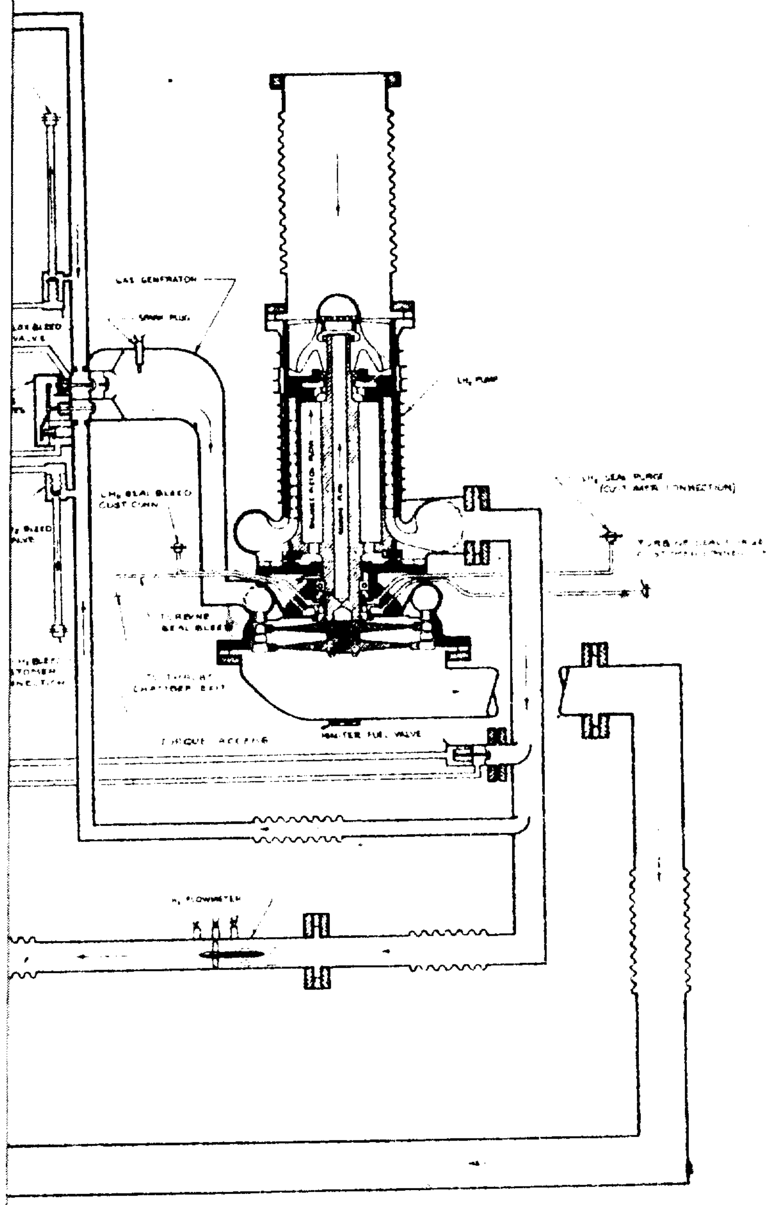

Figure 4.2. J-2 Schematic Diagram

FOLDOUT FRAME 3

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hydrogen valves is also energized. The main hydrogen valve closing control pressure is vented and opening control pressure becomes available to both the ASI oxygen valve and main hydrogen valve.

The main hydrogen valve opens to its first position and liquid hydrogen flows through the thrust chamber jacket, through the injector, and out of the thrust chamber. The ASI oxygen valve also opens and oxidizer flows to the thrust chamber ASI. When the main hydrogen valve reaches its first open position, ports integral with its actuating piston open and allow opening control gas to flow to the gas generator valve and the ASI hydrogen valve.

The gas generator valve opens allowing flow (under tank-head pressure) of liquid hydrogen and liquid oxygen to the gas generator (continued application of closing control pressure causes valve to allow for a hydrogen lead into the combustor). Gas generator and thrust chamber ignition is established.

The function of the gas generator oxygen transition valve, which is closed throughout this period, is to limit temperature excursions and power buildup in the gas generator. This is accomplished by allowing an orificed flow through the valve while it is in the full closed position.

Hot gas from the gas generator accelerates the propellant pumps, increases the propellant discharge pressures, and provides hydraulic power to the gimbaling system. The power output of the gas generator remains constant, and the propellant pump discharge pressures stabilize.

A pressure switch, located in the ASI oxidizer feed line, senses the increased turbopump discharge pressure and mainstage signal is initiated (common signal to all clustered engines) This signal energizes the main oxygen valve control valve and allows opening control gas to flow to the main oxygen and main hydrogen propellant valves, and the gas generator oxidizer transition valve Simultaneously, closing control pressure is vented from the gas generator oxygen transition valve, the gas generator valve, and the main oxygen valve. Opening of the gas generator oxygen transition valve, which is controlled by internally restricting the venting of the pneumatic closing control pressure, controls the temperature excursions and power buildup of the gas generator through transition and into mainstage.

The main oxygen valve opens in approximately 300 msec and oxygen flows into the combustion chamber, the main hydrogen valve travels to its full open position (125 msec). The engine spark ignition system is de-energized by a timing circuit 500 msec after the oxidizer valve control valve is energized. The helium purge gas to the thrust chamber oxidizer side is checked out by the higher combustion pressure of the thrust chamber as smooth transition into mainstage operation occurs.

Gas pressurization for the vehicle fuel tank is supplied from the thrust chamber hydrogen injector manifold. A check valve is installed in the line to prevent flow of fuel from the vehicle tank to the thrust chamber assembly prior to engine start A diaphragm, designed to relieve at pressures attained during the transition phase, is also provided to guarantee against leakage into the thrust chamber.

Gas pressurization for the vehicle oxidizer tank is supplied from a heat exchanger located in the oxidizer pump turbine exhaust duct utilizing oxygen bled from the high pressure propellant duct.

Restart

The J-2 engine is designed for multiple restart. Restart, or multiple start, capability is inherent in the electrical ignition system for thrust chamber and gas generator ignition. The restart capability is enhanced by a simplified start technique which utilizes main propellant tank pressures. Restart capability has also been designed into the pneumatic supply system. The helium sphere is sized for two restarts under temperature and pressure restraints as presented in section 7.

Starting Thrust

The instantaneous rate of thrust increase will not exceed 400 lb/msec with maximum overshoot on thrust buildup not exceeding 3 percent, as shown in Fig. 4.3. The comparable thrust buildup at sea level is shown in Fig. 4.4.

Propellant Consumption

During starting transient it is estimated that 120 lb of propellants will be consumed in the period from initial start signal until 90 percent thrust is attained.

Cutoff Sequence

The cutoff signal, received from the vehicle programmer, energizes the engine-secured timer, and de-energizes the propellant valve control valves.

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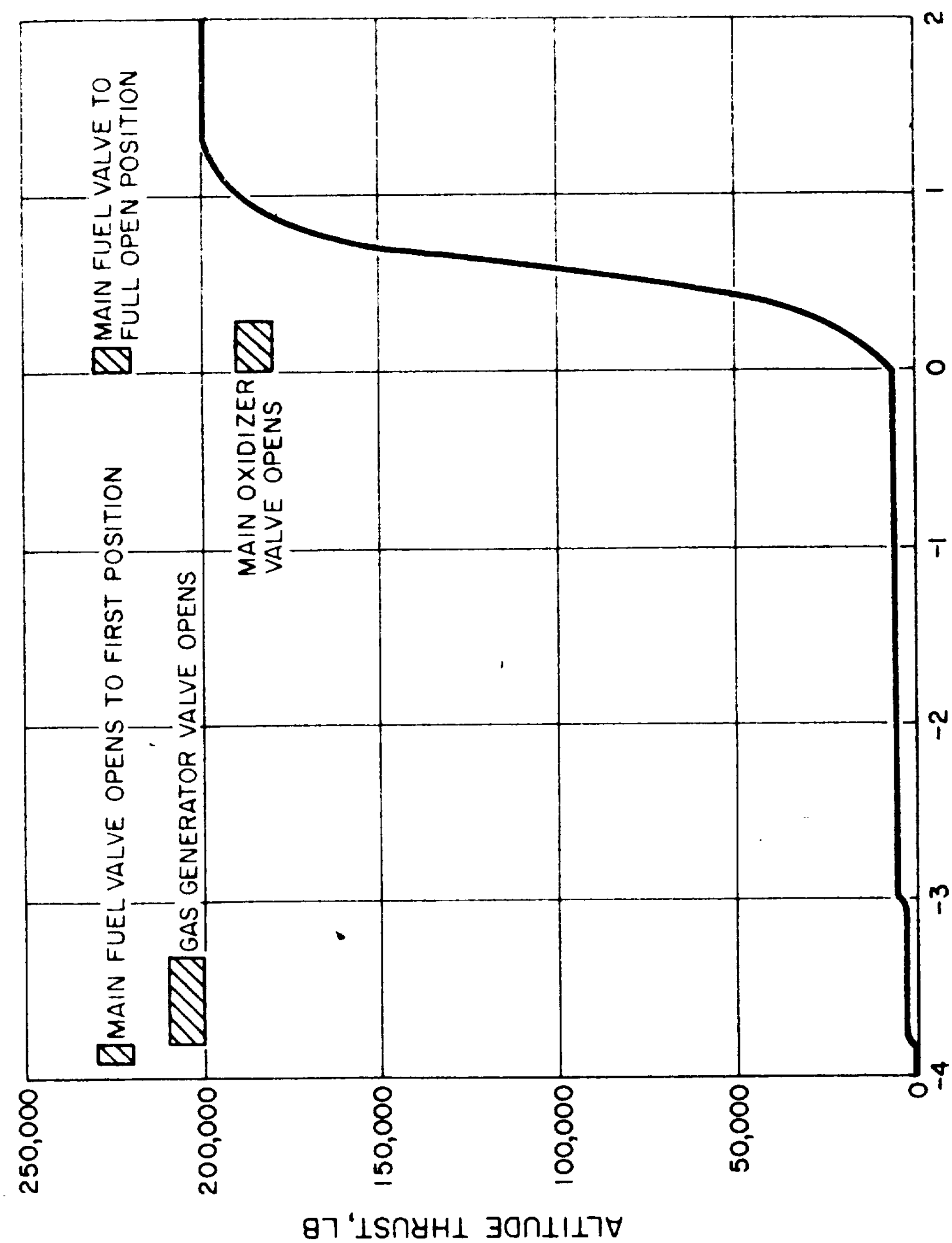


Figure 4.3. Estimated J-2 Thrust Buildup (altitude start)

[redacted]

[redacted]

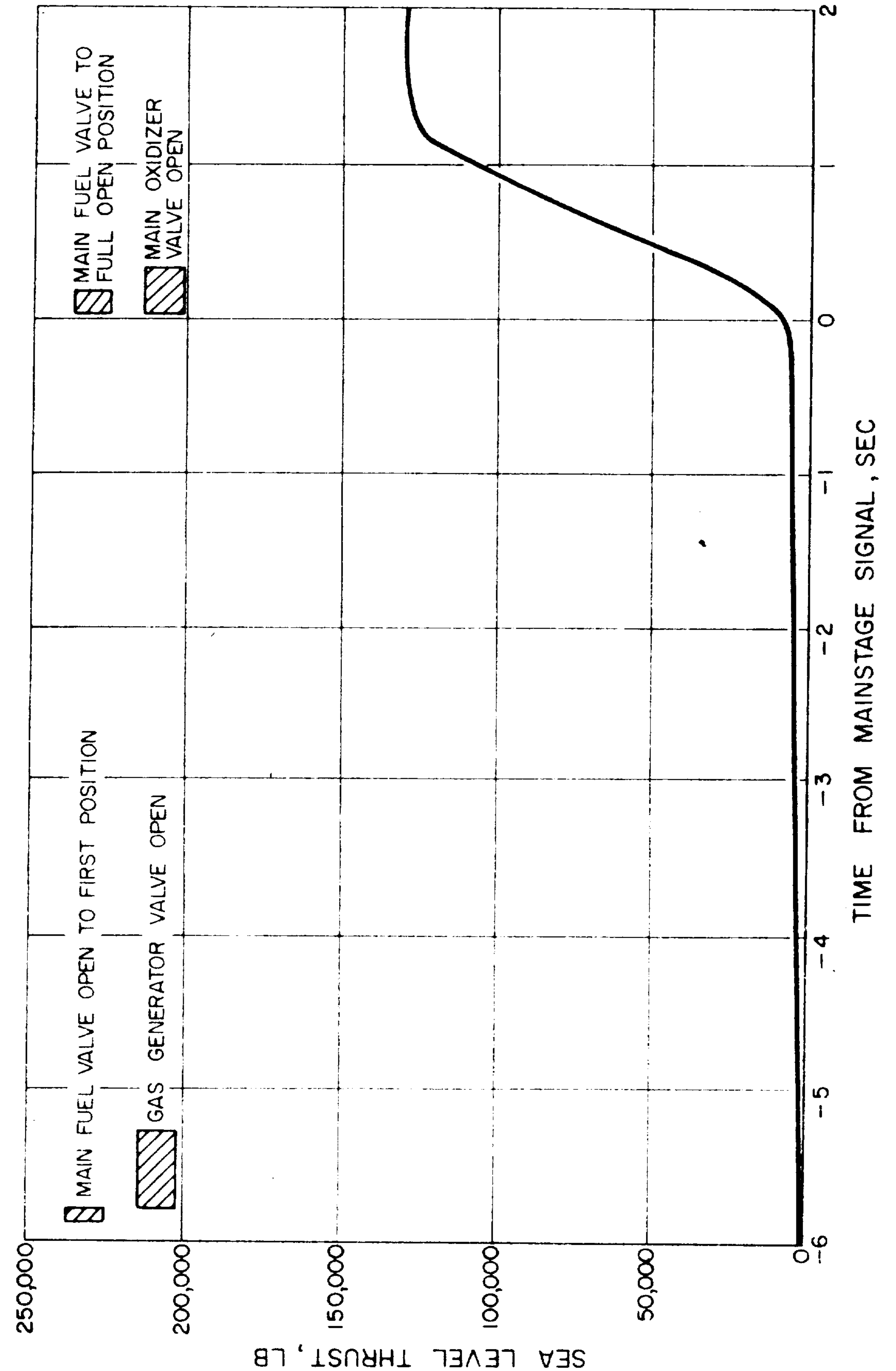


Figure 4.4. Estimated Thrust Buildup at Sea Level

[redacted]

The gas generator valve opening control gas flows through a vent line to the vent port of the main fuel valve control valve. A check valve is located in this vent line to prevent main fuel valve opening control gas from circumventing the safety interlock of the fuel valve and opening the gas generator valve prematurely. The opening control pressure to the ASI oxygen valve also vents through this line. The opening control pressure for the ASI hydrogen valve and the main hydrogen valve (first position) is also vented through this control valve. Simultaneously, closing control pressure is applied to the main hydrogen valve and to the gas generator valve (oxidizer side only). Closing control pressure is supplied to the oxidizer side of the gas generator valve only, to stop the flow of oxidizer as quickly as possible and reduce the power to the turbine, thereby minimizing the cutoff impulse. Closing control pressure is also applied to the main oxidizer valve and to the gas generator oxidizer transition valve. The control lines of the respective propellant valves are restricted to provide for suitable cutoff impulse and a fuel rich cutoff. At this point all propellant flow ceases and the engine thrust decays.

When the combustion pressure in the thrust chamber has decayed sufficiently, the purge line check valve opens and purge gas flows to the oxidizer dome. The engine secured timer de-energizes approximately 500 msec after the cutoff signal from the vehicle programmer, and the helium supply solenoid control valve drops out. The supply gas is cut off by the regulator closure, purge gas stops flowing to the thrust chamber dome, closing control pressure on the gas generator propellant bleed valves is vented, and the bleed valves open.

Cutoff Impulse

The cutoff impulse, measured from receipt of an electrical cutoff signal at the rocket engine, will not vary more than ±1500 lb-sec from an established nominal and will not exceed 50,000 lb-sec. Estimated thrust decay is presented in Fig. 4.5.

Caution is necessary when using the thrust decay curve of Fig. 4.4 in the region of thrust tailoff. From flight test data with liquid oxygen/RP-1 engines, the thrust tailoff is pronounced and exists for a period of approximately 2 sec beyond the estimated thrust decay characteristics for liquid oxygen/RP-1. It is recommended that staging sequencing and thrust loading should not be specifically designed upon the estimated thrust tailoff characteristics presented in Fig. 4.4 until J-2 engines experimental data are available.

Propellant Consumption During Cutoff

It is estimated that 100 lb of propellants will be consumed during the transient period from engine cutoff signal to zero thrust.

Turbopump Speed Decay at Shutoff

It is estimated that after engine shutoff signal, the fuel turbopump speed will decline from the nominal mainstage speed of 26,435 rpm to 0 rpm in 10 sec. The estimated decline of the oxidizer pump under the same conditions is from 8950 rpm to 0 rpm in 4 sec. The estimated speed decay is based upon the presence of propellants, as liquid, within the turbopump casings at cutoff. In the absence of viscous drag due to the propellants, the time for speed decay would be considerably increased.

[redacted]

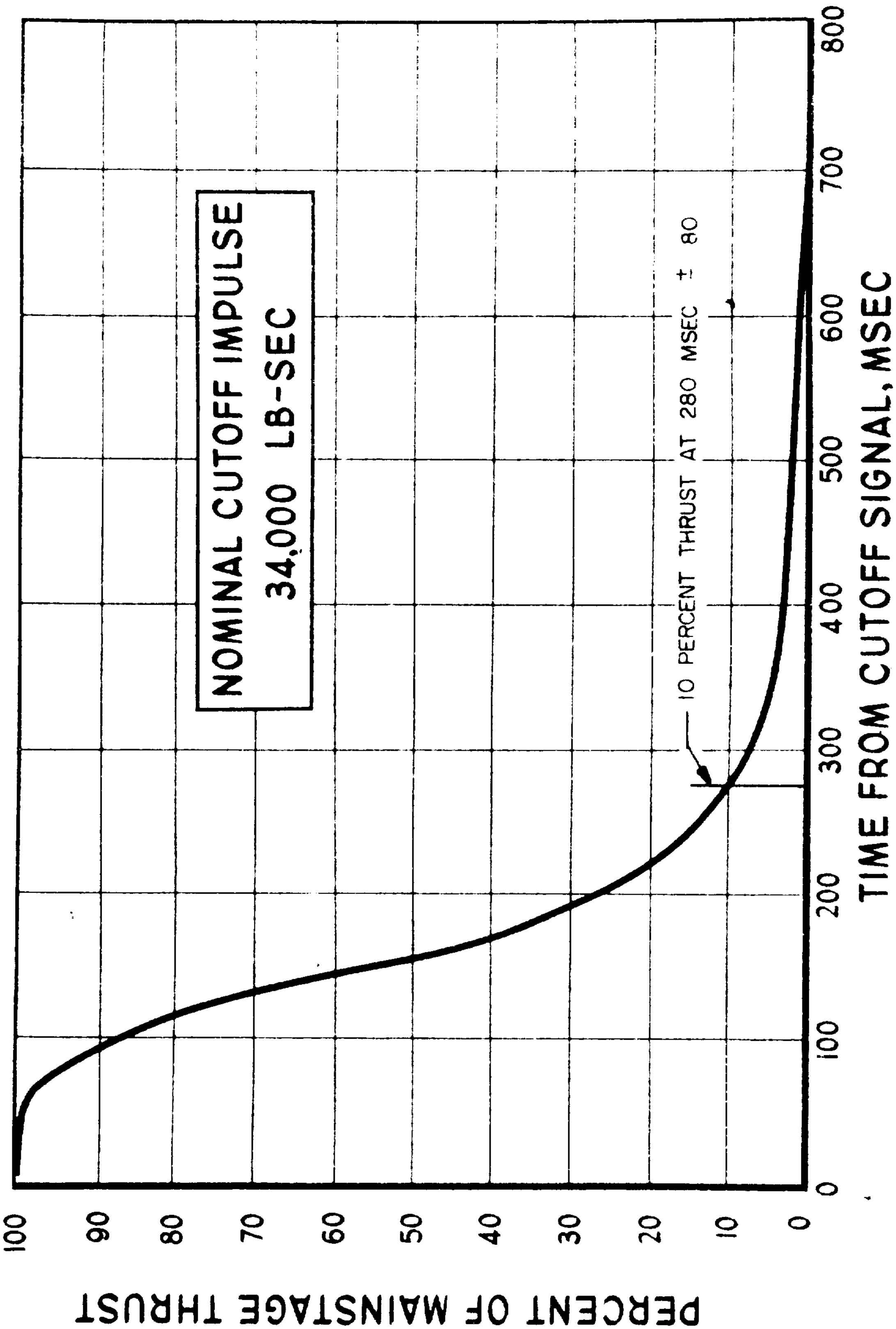


Figure 4.5. Estimated J-2 Thrust Decay

[redacted]

WEIGHT AND INERTIA

WEIGHT AND INERTIA DATA

J-2 ENGINE WEIGHT AND GIMBALED MASS VALUES

The J-2 engine weight, along with gimbaled mass values, and a sketch showing gimbal axes are shown in Table 5.1.

GYROSCOPIC MOMENT AND MOMENT OF INERTIA

The gyroscopic moment, and moment of inertia data, are as follows:

1. Total gyroscopic moment, 928 ft lb/radian/sec attitude change (required to effect an engine attitude change)
2. Net gyroscopic moment, 290 ft lb/radian/sec attitude change (moments at 90 deg to direction of engine attitude change)
3. Fuel turbopump gyroscopic moment, 609 ft lb/radian/sec
4. Oxidizer turbopump gyroscopic moment 319.0 ft lb/radian/sec

The above values are based on the following factors:

1. Fuel turbopump mainstage speed, 26,435 rpm (clockwise looking aft)
2. Oxidizer turbopump mainstage speed, 8950 rpm (counterclockwise looking aft)

TABLE 3.1

J-2 WEIGHT, BALANCE, AND INERTIA

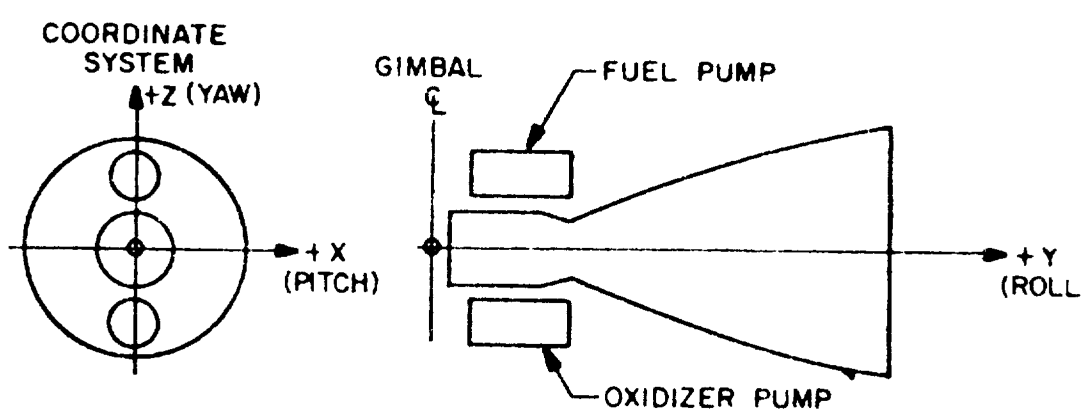
	Weight, lb	Center of Gravity, in.			Moment of Inertia* slug ft^2		
		$\bar{y}$	$\bar{x}$	$\bar{z}$	I_{yy}	I_{xx}	I_{zz}
Engine							
Dry	2038	29.5	3.2	0.9	259*	617*	573*
Burnout	2132	28.8	3.3	0.2	272*	649*	597*
Wet	2165	28.5	3.2	0.2	273*	653*	602*
Wet Gimbaled Mass	2067	30.1	3.4	0.5	270**	1100**	1064**

K = radius of gyration

$$K = \sqrt{\frac{I}{M}}$$

*Moment of inertia about specified center of gravity

**Moment of inertia about specified gimbaled axis



NOTE: Positive Directions Indicated by Arrowheads

3. Fuel turbopump polar moment of inertia, 0.220 slug ft^2 (wet rotating mass)
4. Oxidizer turbopump polar moment of inertia, 0.341 slug ft^2 (wet rotating mass)

It will be noted that the fuel and the oxidizer pump rotate in opposite directions tending to reduce the total precession moment below the total gyroscopic moment required to effect an engine attitude change at a given rate.

The estimated time for turbopump stoppage after engine cutoff signal is discussed in section 4.

MOUNTING

THRUST CONTROL (GIMBAL) SYSTEM

GIMBAL BLOCK

The J-2 engine basic mount consists of a plain spherical-type gimbal bearing to transmit engine thrust to the vehicle. Engine torque and hanging loads are transferred by a bar extending through the sphere center. A teflon fiberglass composition coating provides dry low-friction bearing surfaces which are capable of operating in a vacuum at design temperature limits.

Thrust is transmitted to the space frame structure through the trunion spherical socket. The socket is contained in a heat-treated metal block with two crossed keys set in a plane normal to the thrust and on the vehicle side of the block. The keys absorb loads normal to the center-line of thrust and torque loads imposed by the gimbaling actuators and the vehicle gyrations. The thrust transmitting bearing surface and attaching bolt holes lie within the quadrants between the keys. In attaching the gimbal block to the stage structure a minimum of 90 percent of the bearing surface should be utilized.

Gimbal blocks may be attached to the vehicle structure either by use of bolts or by cap screws installed with heads on the engine side. A detail of the gimbal block is shown in Fig. 2.1.

Loads

Maximum loads which will be transmitted to the vehicle frame through the gimbal block during flight are as follows:

Lineal Forces (KIP)			Moments (KIP IN.)		
Fx	Fy	Fz	Mx	My	Mz
±58	-302 + 90	±60	±285	±152	±269

Table 5.1, section 5, presents parameters for coordinate system orientation.

BEARING DUST AND MOISTURE PROTECTION

A gimbal boot has been provided to protect the bearing surfaces from dust, water, and foreign materials. The boot has a bellows configuration with the convolutions permitting the required gimbaling motion. A silicone impregnated fiberglass material is used for the bellows section, and molded silicone rubber seals at the two attach points. Both gimbal boot attach points are on the gimbal assembly. This results in an integral gimbal and boot assembly. Provisions have been made to permit installation or removal of the boot on the assembled engine with a minimum of effort.

GIMBALING SYSTEM

ACTUATOR ATTACH POINTS

In addition to basic support from the gimbal block, the engine is supported by two gimbal actuators attached to the thrust chamber dome, each 90 deg to the other with respect to the centerline of thrust. A detail of the gimbal attachment is shown in Fig. 2.1. The dome attach points are located on a radius of 11.875 in. from the thrust centerline, and provide for 1-in.-dia shear bolts or pins to be installed in double shear through a spherical alignment bearing. Orientation and details of the mounting attach points are shown in Fig. 2.1. The actuation envelope and installation requirements are described in Fig. 6.1. The two actuators are not supplied with the J-2 engine.

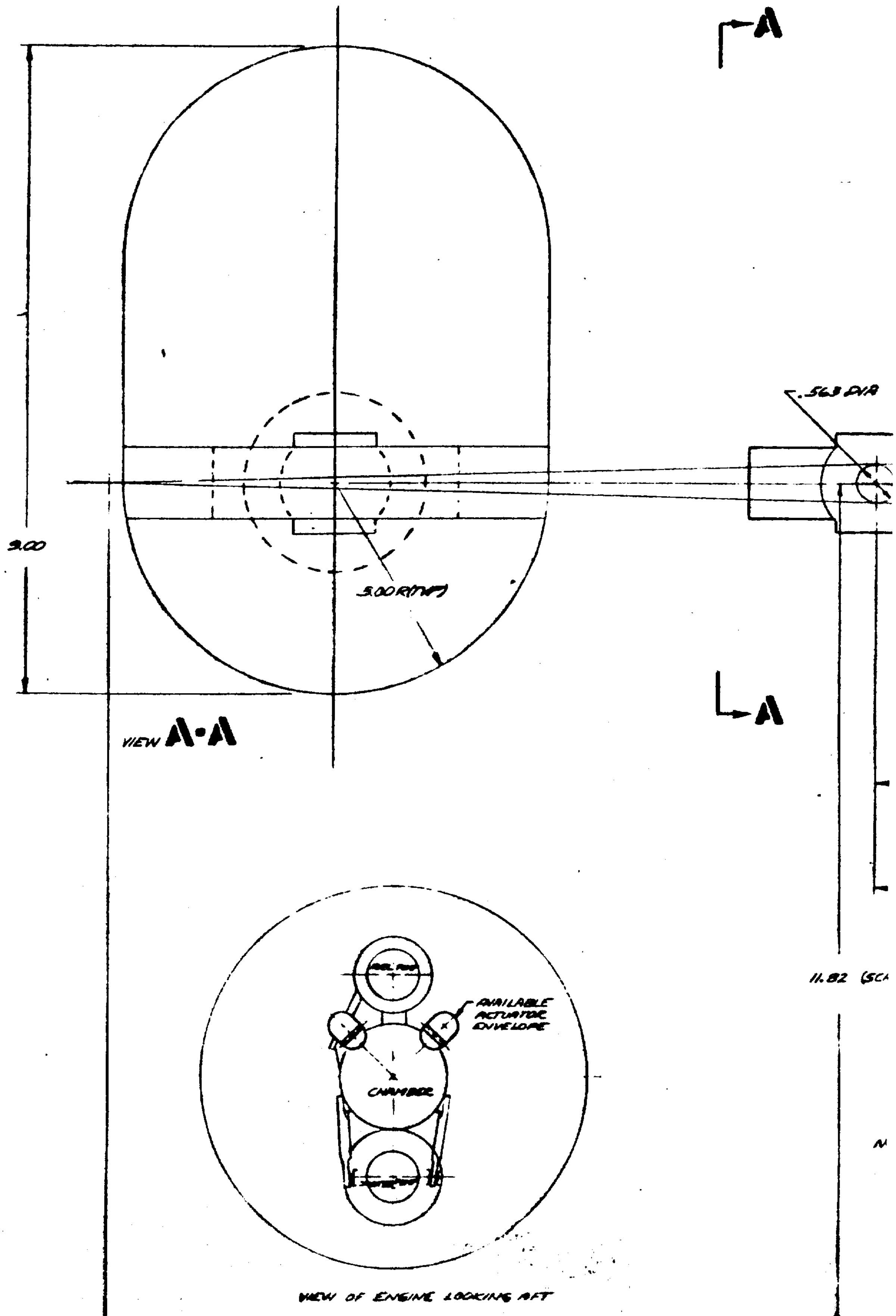
MOVEMENT

The engine is designed to withstand a 7-deg functional gimbal angle simultaneously or individually from each actuator, plus a 1/2-deg actuator overtravel position-stop allowance. The resultant corner position allowance is for approximately 10 deg 25 min gimbaling. Overtravel positive stop provisions must be integral with the actuator.

The structural design of the engine permits functional gimbaling of 7 deg only, as measured from the vehicle flight axis, in any plane or any combination.

Forces transmitted to the gimbal block resulting from gimbal actuator forces are presented in a previous paragraph of this section.

[redacted]

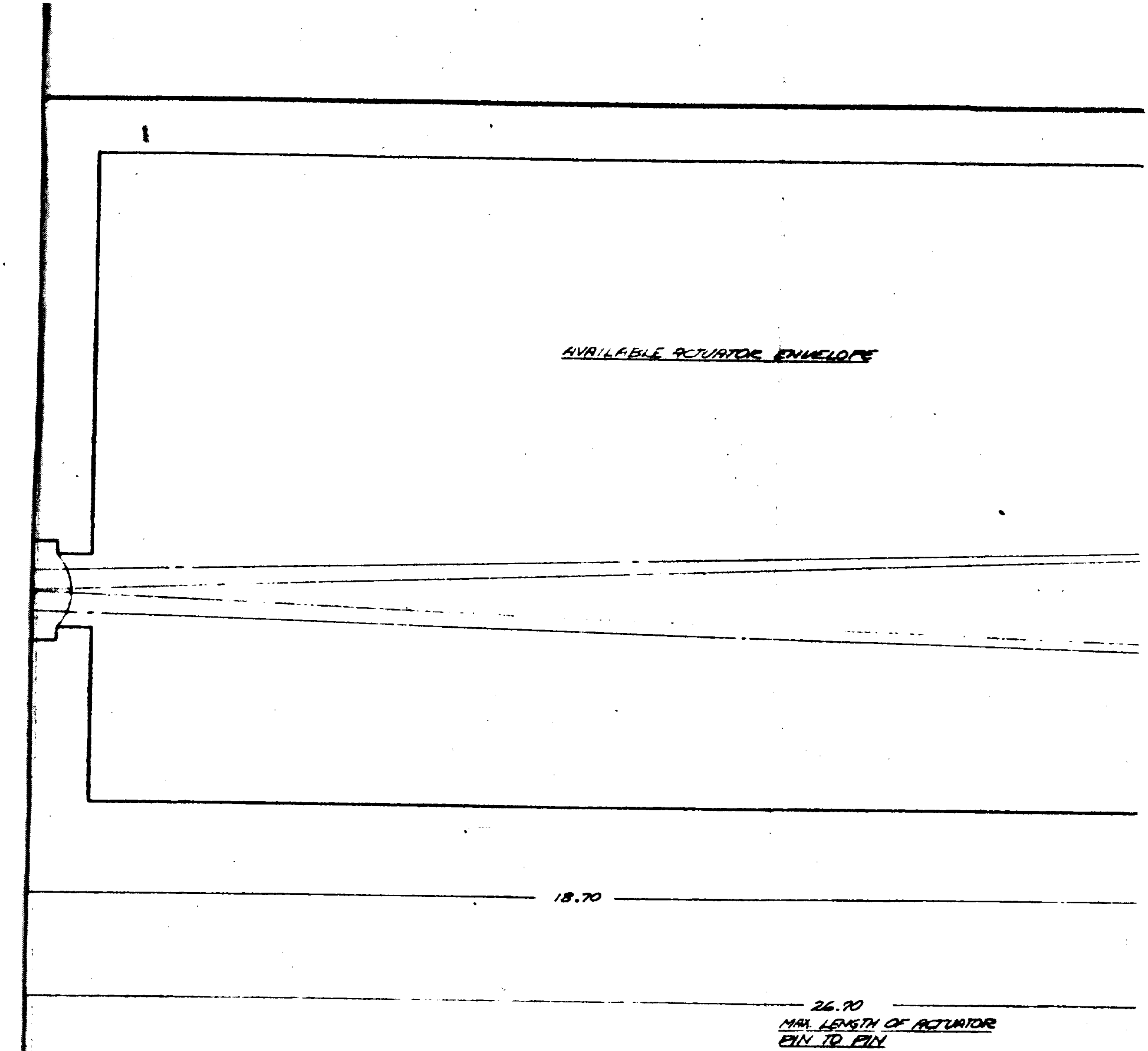


R-2661-4P

FOLDOUT FRAME

FORM R 198-E-13

DIETERICH-POST CLEARPRINT 1000H



LED)

MAXIMUM GIMBAL ACTUATOR LENGTH IS ESTABLISED BY THE FOLLOWING DATA:

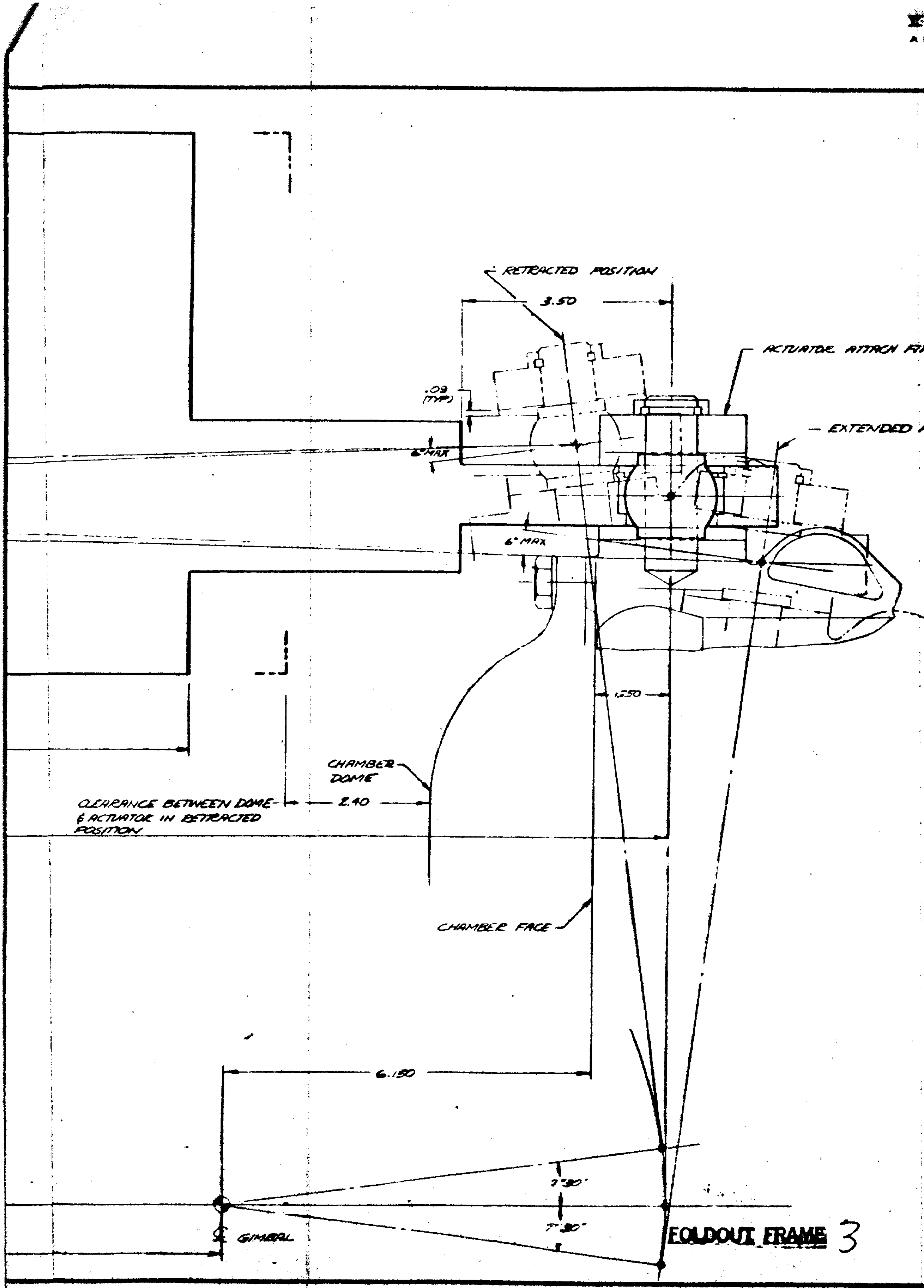
1. MAX. GIMBAL ANGLE ~ 7°30'
2. ACTUATOR MOMENT ARM FROM ℄ OF ENGINE ~ 11.875.
3. MAX. ANGULARITY OF ACTUATOR BLADE FITTING IN CHAMBER ATTACH FITTING ~ 6°.
4. THRUST MISALIGNMENT CAPABILITY ~ ± .25.
5. AIRFRAME ATTACH POINT TOL. IN RELATION TO GIMBAL CENTER ~ ± .030 (ASSUMED).

OTE: LONGER ACTUATOR THAN THAT SHOWN AS MAX. ASSUMING SAME TOLERANCES AS THOSE IN NOTE 4, & 5 REDUCES CHAMBER FITTING CLEARANCES. SHORTER ACTUATOR THAN THAT SHOWN AS MAX INCREASES ALLOWED TOLERANCES SPECIFIED IN NOTE 5 BY INCREASING ACTUATOR TO CHAMBER FITTING CLEARANCES.

ENGINE ℄

29.92
(SCALED)

FOLDOUT FRAME 2



FOLDOUT FRAME 3

ONG

SITION

Figure 6.1. J-2 Envelope Gimbal Actuators

FOLDOUT FRAME 4

ACCELERATION AND VELOCITY

The engine can withstand a maximum gimbaling angular acceleration of 76 rad/sec^2. To achieve a maximum angular velocity of 30 deg/sec, with a sinusoidal variation, an available gimbal acceleration of 2.25 rad/sec^2 is required. The acceleration value is based on a single-plane-actuated simple-harmonic gimbal cycle through a 14-deg displacement angle from one extreme to the opposite, and the maximum gimbal angle velocity of 30 deg/sec occurring as the thrust chamber passes through the zero gimbal angle position. The gimbaling angular acceleration is the total engine angular acceleration allowed. Angular acceleration of the vehicle frame is considered as lateral accelerations applied to the engine, and to fall within the specified lateral acceleration limits.

ACCELERATIONS (TRANSLATING)

With zero thrust from the J-2 engine, the engine will withstand a maximum forward acceleration of 10 g, together with 0.5 g lateral acceleration, or 5 g forward acceleration together with 1 g lateral acceleration.

For the conditions of accelerations resulting from J-2 engine thrust, the limits of combined forward and lateral accelerations for which the engine is designed, are shown in Fig. 6.2.

GIMBALING AND VEHICLE LOADS

During the J-2 engine firing stage the engine is designed to gimbal as much as 2 deg while the vehicle accelerates up to the designed limit of engine forward acceleration. The designed limit of forward acceleration

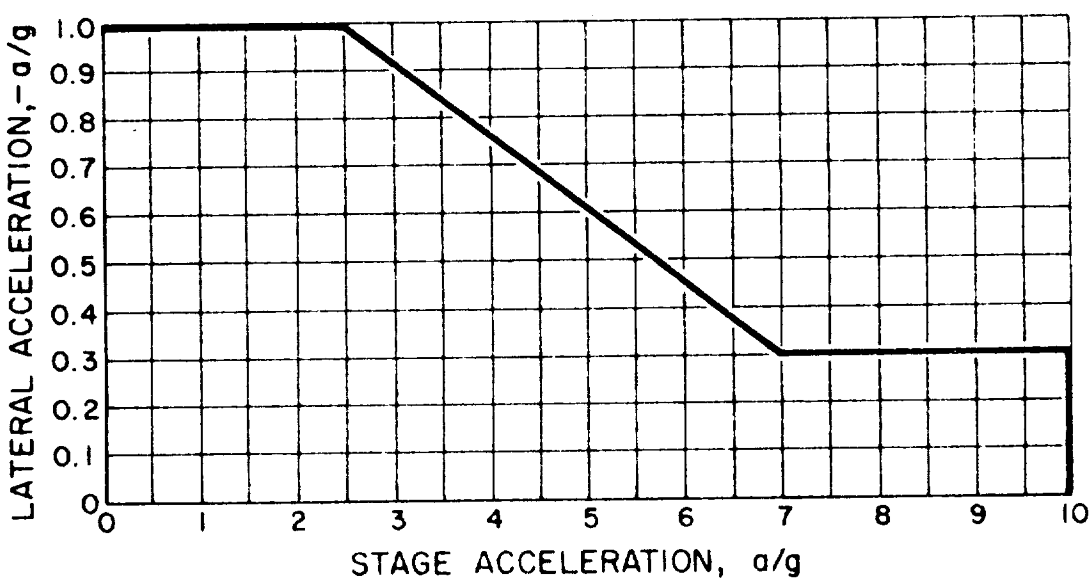


Figure 6.2. Lateral Acceleration vs Stage Acceleration

with respect to engine gimbal angle is shown in Fig. 6.3. These limits occur simultaneously with the lateral acceleration relationship to forward acceleration shown in Fig. 6.2.

THRUST ALIGNMENT

The engine gimbal bearing assembly is capable of adjusting for lateral displacement of the geometric thrust vector of 0.250 in. in any direction. The dynamic thrust vector may deviate from the geometric thrust vector, angularly as much as 0 deg 30 min, and laterally as much as 0.250 in. in any direction, measured at the gimbal axis plane.

ACTUATOR LOADS

The maximum actuator load capacity which may be tolerated by the engine mounting structure is 42,000 lb each.

The minimum actuator loading allowance which should be made to permit operation within model specification limits is 28,500 lb each.

The above given actuator load is based on the following factors:

1.	Longitudinal acceleration, g	2.5
2.	Lateral acceleration, g	1
3.	Angular acceleration, rad/sec^2	2.25
4.	Gimbal bearing friction, in. lb	76,300*

*All moments about gimbal center, moment arm 11.875 in.

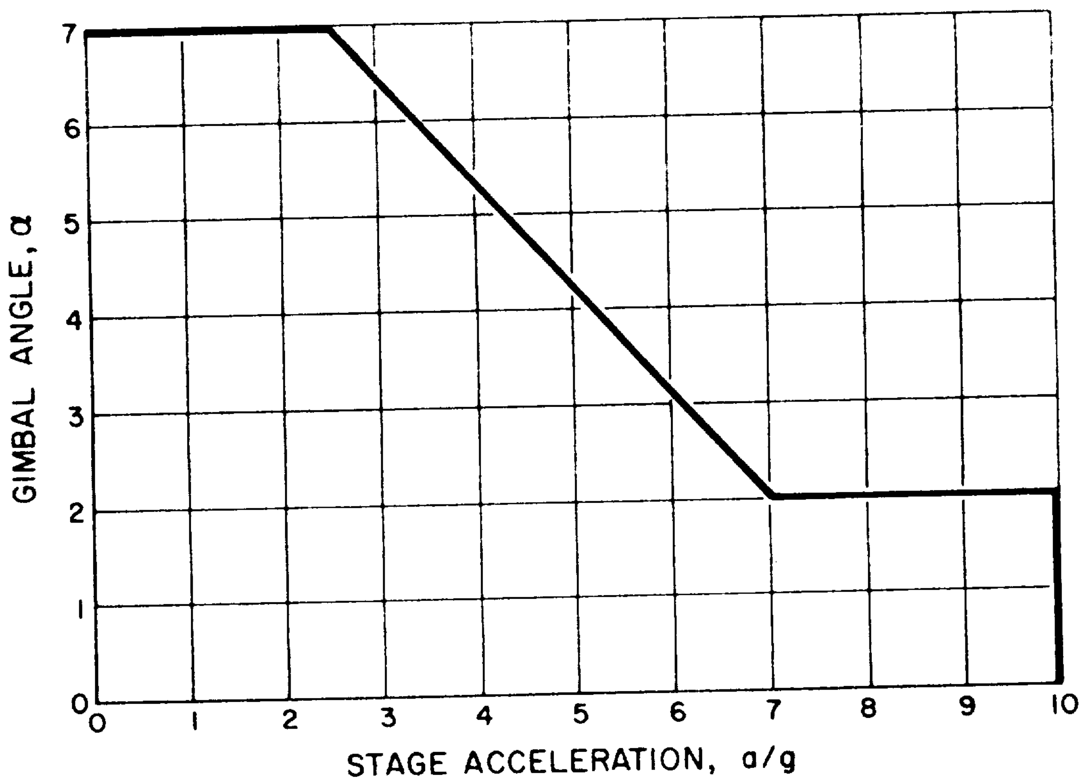


Figure 6.3. Gimbal Angle vs Stage Acceleration

5.	Gyroscopic load, in. lb at 30 deg/sec	5700*
6.	Thrust chamber misalignment, in displacement	0.25
7.	Propellant inlet loading	
	a. Fluid static pressure, in. lb	15,200*
	b. Flex duct axial spring load, in. lb	27,300*
	c. Flex duct bending spring load, in. lb	1400

AERODYNAMIC LOADING

Although the aerodynamic load carrying capability of the engine will be influenced by the way the engine is oriented relative to the vehicle, preliminary information based on a particular configuration may be useful in the early stages of vehicle design. Therefore, the aerodynamic load carrying capability of the engine has been investigated for a particular installation in a clustered configuration with the oxidizer and fuel inlets located on a radial line in the vehicle and the fuel pump outboard. If the +Z axis shown in sketch in Table 5.1 is outboard in a radial direction, the largest deflection of the engine into the slip-stream will occur with the maximum pitch angle of approximately 10 deg about the X axis. In this position the two 42,000 lb actuators have a capability to resist an aerodynamic moment of 300,000 in. lb about the X gimbal axis. It is presumed that this moment results from the impingement of air upon the aft portion of the thrust chamber. With

*All moments about gimbal center, moment arm 11.875 in.

the present design of the thrust chamber no difficulty would be encountered in sustaining both the dynamic pressure (approximately 430 lb/ft^2) and the bending moments encountered if the total moment of the aerodynamic load does not exceed 300,000 in. lb.

Engine plumbing above a plane 77 in. aft of the gimbal center has not been designed to withstand airflow, as it is not expected to extend into the airstream.

PROPELLANT SYSTEMS

PROPELLANTS AND PRESSURANTS

PROPELLANT AND PRESSURANT SPECIFICATIONS

The following specifications are applicable to the propellants and pressurants employed in the J-2 engine.

Liquid Oxygen	MIL-P-25508B	Dated 22 June 1960
Liquid Hydrogen	MIL-P-27201	Dated 21 May 1959
Gaseous Helium	Bureau of Mines	Grade A

HELIUM

The moisture content of the helium employed in the J-2 pneumatic system is extremely critical because of the low temperature environment resulting from the use of liquid oxygen and liquid hydrogen as propellants. A helium dew point maximum of -100 F, at atmospheric pressure, is a requirement. Foreign matter in the helium must be limited to a maximum of 0.01 mg/liter at 60 F and 14.7 psia. Helium supplied to the engine pneumatic supply sphere must pass through a 10-micron nominal filter having a 25-micron absolute rating. The helium supply sphere is sized to hold enough gaseous helium for three starts (two restarts) if charged to 4500 psia at -100 F. Less helium may be loaded if only one start is desired as shown in Fig. 7.1.

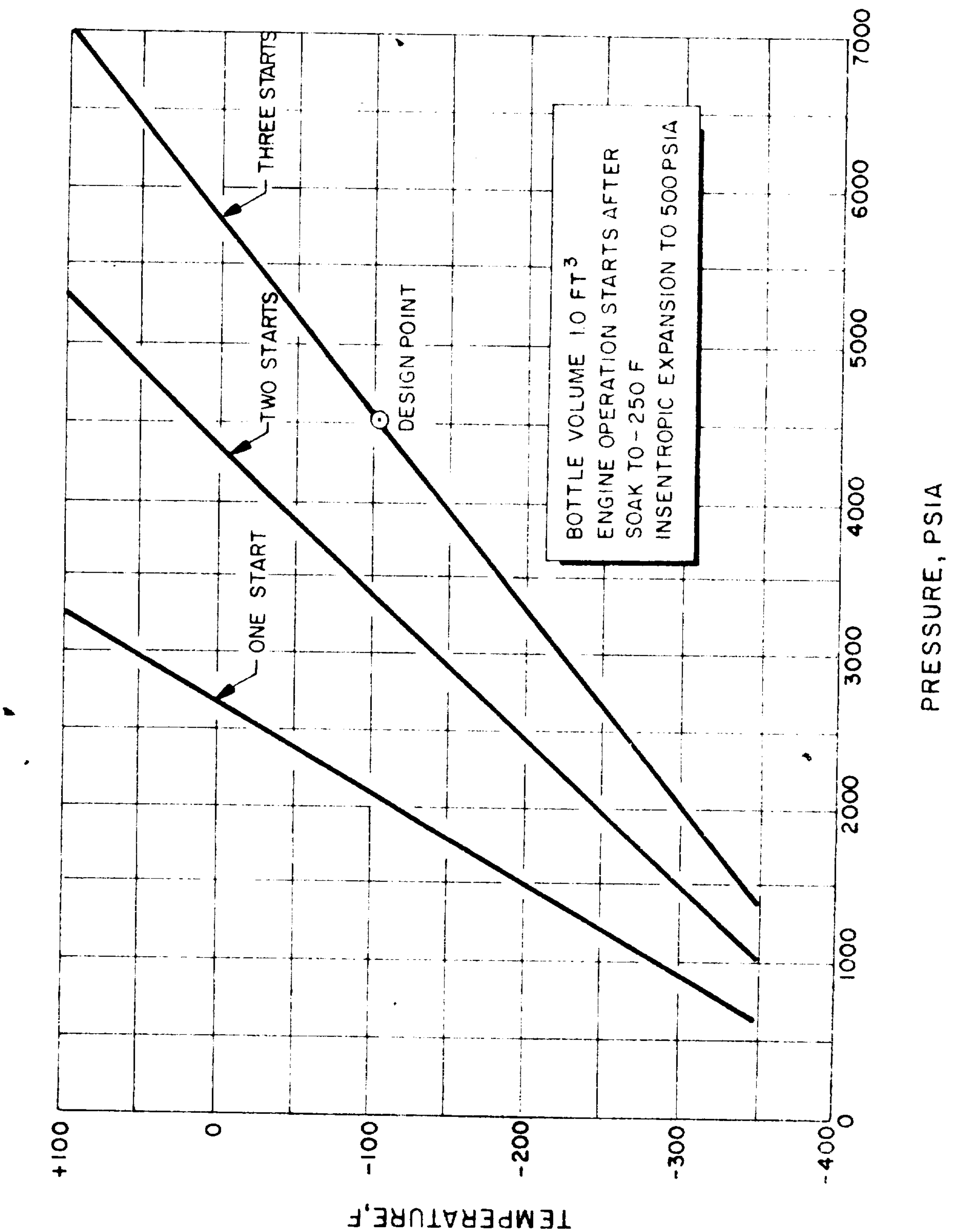


Figure 7.1. Number of Start Cycles as a Function of Helium Temperature and Program Requirements at Loading

PROPELLANT PUMP INLET DUCTS

PROPELLANT DUCT CONNECTION FLANGE LOADS

The J-2 engine propellant inlet ducts connect to the vehicle frame feed system ducts by bolted flange fittings. The two engine propellant ducts are of flexible metal construction and are identical. Each duct consists of two convoluted bending sections restrained by gimbal rings, one at each end of the duct, and one convoluted central member designed to absorb axial motion during engine gimbaling. When under pressure, the engine mounted flexible section is dependent upon the vehicle duct flange for axial restraint. Accordingly, the vehicle duct flange at the engine connection must be rigidly oriented with respect to the X, Y, and Z axes.

Propellant pressurization and forces required to deflect the convolutions during 10 deg corner gimbaling result in the following maximum forces imposed on the vehicle structure.

	Lineal Forces (KIP)			Moments (KIP-in.)		
	Fx	Fy	Fz	Mx	My	Mz
Oxidizer Inlet Line	±2.1	+2.4 -13	±2.1	±6.4	±46.5	±6.4
Fuel Inlet Line	±1.2	+2.4 -8.1	±1.2	±2.7	±46.3	±2.7

Table 5.1 presents parameters for coordinate system orientation.

The axial spring rate of an unpressurized inlet line is approximately 215 lb/in. The bending spring rate for vehicle connecting end flange tilt is 4.75 ft-lb/deg about a point 2 in. below the vehicle flange. For the engine connecting end it is 5.42 ft-lb/deg about a point normally 2 in. below the vehicle flange. The inlet line customer connect flanges are dimensioned to a nominal free position. The above spring rate may be used to determine additional forces imposed by axial position change of the flange.

DUCT LOCATION TOLERANCE

Vehicle flanges must be positioned and during engine operation remain within the following tolerance with respect to the engine flange position defined on the customer connect drawing:

1. Axial location (y axis): ±0.30 in.
2. Transverse location (xz plane): ±0.08 in. in any direction
3. Angular misalignment (out of xz plane): ±1/2 deg
4. Misalignment of bolt pattern: As the duct is centered by the attachment bolts, the bolt pattern is required to be concentric with the inside diameter.
5. Rotational alignment of bolting flanges (about the y axis): The vehicle connecting duct must be provided with bolt holes to allow for adjustment of a minimum of ±1 deg angular misalignment. During operation, rotational movement of the vehicle duct flange about the duct centerline shall be limited to 0.2 deg relative to a plane through the pump axis and the engine centerline.

The engine inlet flexible ducts are designed to withstand 500 gimbal movement cycles, under the most adverse installation conditions listed here, when the ducts are at design operating pressure. Under zero pressure, the ducts will withstand 3000 cycles without fatigue failure.

A movement cycle is defined as the required duct deflection from its neutral position to the maximum movement and return to neutral.

ENGINE CONNECTING PROPELLANT DUCT SEAL

Propellant duct flanges are designed for use with a Naflex-type self-energizing double seal. This seal is designed expressly for the joint, and has been tested to demonstrate its suitability while subjected to the transverse and axial loadings to be encountered during engine gimbaling. It should be noted that the thickness of the seal is not included in the inlet duct lengths shown in Fig. 2.1.

PROPELLANT SYSTEM PRESSURES

FUEL AND OXIDIZER SYSTEMS

The engine fuel system is calibrated to a 25 psia nominal mainstage total inlet pressure, and is designed to operate with a minimum NPSH of 130 ft (4 psi) at the pump inlet. (NPSH is defined as the difference between the pump inlet total pressure-head and the fluid vapor pressure-head.) The engine oxidizer system is calibrated to a 32 psia nominal mainstage total inlet pressure, and is designed to operate with a minimum NPSH of 25 ft (12.5 psi) at the pump inlet. Reduction of static fuel pressure at the beginning of the start cycle will adversely effect the tank-head-start characteristics. An increase of static pressure will cause excessive duct stresses during engine shutdown surges. The flexible duct sections supplied as part of the engine propellant system will each add a pressure drop to the vehicle propellant feed system and therefore, must be considered by the vehicle manufacturer, in designing, to maintain the above minimum NPSH values.

The pressure drop for the fuel pump inlet duct when in the nominal position is 0.5 psi at the nominal flowrate. At the same flowrate, with the duct fully extended and offset, the pressure drop is 1 psi.

Pressure drop for the oxidizer pump inlet duct when in the nominal position is 0.5 psi at the nominal flowrate. At the same flowrate, with the duct fully extended and offset, the pressure drop is 2.0 psi.

SURGE AND PUMP INLET PRESSURE

The engine inlet ducting, turbopump inlet casing, and turbopump mounting structure of the J-2 engine are designed to withstand surge pressures of 100 psi above the nominal inlet pressure in the oxidizer system and 50 psi in the fuel system. In terms of total pressure (surge plus nominal operating), the limitations are 75 psia for the fuel inlet system and 132 psia for the oxidizer inlet system. It is therefore necessary, in every specific application of the J-2 engine, to evaluate possible vehicle ducting configurations in terms of elevation from engine connect point to propellant tank discharge and of ducting-run length and diameter. Elevations of from 20 to 30 ft coupled with vehicle terminal effective fluid accelerations of 4 to 5 g result in significant static pressures with a fluid as dense as liquid oxygen. The values of ducting-run length and diameter are of importance in evaluating pressure surges which are acoustical phenomena.

Rocketdyne has developed a digital computer program for analyzing waterhammer effects based on a solution to the one-dimensional wave equation. This program has significant success in predicting pressure surging caused by rapid valve closure in series piping systems. The input data for this program are diameter, run length, wall thickness, modulus of elasticity, and Poisson's ratio for the ducting; the free acoustic velocity, adiabatic bulk modulus, and flowrate of the propellant; and the propellant valve closing characteristic. Any questions concerning proposed vehicle ducting layouts in regard to pressure surges should be referred to Rocketdyne and be accompanied with the aforementioned ducting data.

To minimize cutoff impulse and maintain the specified close tolerances on cutoff impulse variation, the J-2 engine employs rapid main propellant valve closures; this results in waterhammer pressure surges. The nominal closing rates are 100 msec for the main oxidizer valve and 200 msec for the main fuel valve. The system surge pressures have been evaluated for the most rapid closure anticipated under propellant flow and pressure conditions encountered at the maximum of mixture ratio control when using 7-in. ID piping and run lengths of 13 ft for the fuel and 6.5 ft for the oxidizer ducting. As a result of these analyses, the J-2 engine main oxidizer valve, butterfly type, has been designed to follow a linear area decay characteristic on closing to minimize pressure surges caused by rapid deceleration of over 450 lb/sec of liquid oxygen flow. The properties of hydrogen and the value of maximum fuel flowrate were such that the main fuel valve, also a butterfly type, required no modification of the linear position characteristic associated with butterfly-type valves.

The maximum run lengths of propellant feed ducting which generate surges safely below the maximum surge levels, at nominal inlet pressure conditions, are 13 ft for the fuel system and 6.5 ft for the oxidizer system. The maximum run lengths are based on the use of 7-in.-ID low-pressure stainless-steel piping. The utilization of larger diameter ducting will reduce the surge pressure as a result of decreasing the flow velocity; however, the relation is not direct. If structural requirements of proposed vehicles dictate run lengths greater than those mentioned, the investigation of surge suppressing methods is recommended.

STARTING PUMP INLET PRESSURES

The J-2 engine is unique among existing large liquid-propellant rocket engines because of the utilization of the tank-head-start technique. This start technique avoids the use of gas generating spinner cartridges, start tanks, etc., with the attendant complexity and possible limitations on restart capability. Because of the sensitivity of the engine to turbopump inlet pressure levels during the initial starting transient, the following paragraphs are provided to clarify the effects of starting pump inlet pressures while employing the tank-head-start technique.

The J-2 engine program has employed a mathematical lumped-parameter model as a tool to predict system starting transients prior to inception of actual engine development testing. Through this analytical effort, gas generator thermal excursions during the starting transient were thoroughly analyzed with respect to turbopump inlet pressures. The necessity for this analysis is discussed in the following paragraphs.

The control system for gas generator temperature and flowrates consists of two fixed orifices, one each in the oxidizer and fuel lines. At mainstage these orifices are adjusted for a gas generator mixture ratio of approximately one. The pressure drops, including the control orifices, which determine the fuel and oxidizer flowrates into the gas generator are those from the respective pump discharge volutes to the gas generator chamber. At mainstage, the pressure drop across the gas generator oxidizer line is less than that across the hydrogen line, while at start, the nominal oxidizer pump inlet pressure is 32 psia and the fuel pump inlet pressure is 25 psia. Therefore, for any given gas generator chamber pressure, the pressure drop across the gas generator feed lines (employing mainstage control orifices) is higher on the oxidizer side, which means the oxidizer flowrate will be greater than the fuel flowrate. A further tendency toward oxidizer-rich operation results from the ratio of liquid oxygen density to liquid hydrogen density being 16:1. Additionally, the engine starting technique employs a partially opened main fuel valve and a fully closed main oxidizer valve. Following ignition in the gas generator and initial acceleration of both turbopumps, the oxidizer pump discharge pressure increases more rapidly than that of the hydrogen pump because of operation against a closed main valve, and because the oxygen pump design speed is lower than that of the hydrogen pump. These conditions result in temperature excursions in the gas generator.

To control the gas generator temperature excursions, a gas generator oxygen transition valve is located in the gas generator oxidizer feed line. This valve presents a high resistance to oxygen flow prior to opening the main propellant valves. This produces a higher pressure drop in the oxidizer feed line during start and will preclude oxidizer-rich gas generator operation and the resultant high temperatures. Simultaneous with main oxidizer valve opening at the mainstage signal, the transition valve moves at a controlled rate to its fully open position to permit engine operation at the mainstage level.

Prior to initiation of the engine starting phase it is required that the pump inlet pressures be no less than the nominal mainstage values.

PROPELLANT SYSTEM PREVALVES

The J-2 engine has been designed under the assumption that the engine main propellant valves are the only valves in the vehicle propellant feed systems. Additional feed system valves, called prevalves, would only be added to fulfill vehicle requirements independent of the engine. If prevalves are added, and if their cycle of operation is arranged so that both the prevalve and the engine valve could be closed at the same time trapping liquid cryogenic propellant, then the vehicle manufacturer must provide for pressure relief of the space between the valves. Pressure would build up because of propellant thermal expansion. It would also build up because of flexible duct contraction if the engine is gimbaled while in this condition. Total pressure limitation is presented in the previous paragraphs on surge and pump inlet pressure. Thermal expansion which takes place will affect an engine fuel volume of 1.50 cu ft and an oxidizer volume of 1.48 cu ft. Engine gimbaling at the specified rate, with a liquid-filled duct, will require a relief capacity of 2.9 gal/sec.

The two propellants must be in contact with the engine for at least 30 min prior to engine operation, to assure thorough pump childown. The prevalves must be opened 30 min prior to firing if prevalves are used to isolate the engine during extensive holds with tanked propellants.

ELECTRICAL

J-2 ELECTRICAL SYSTEM

The J-2 electrical control system consists of a sequence controller to properly sequence engine start and cutoff, a spark ignition system to establish ignition in the gas generator and thrust chamber, and an ASI pressure switch to sense proper ignition and signal mainstage.

SEQUENCE CONTROLLER

The sequence controller is a complete solid-state design thereby eliminating problems associated with mechanical relay contacts. The controller performs the functions of circuit condition monitoring, protective interlocks on critical circuits, and proper sequencing of the engine during the start, cutoff, and restart phases of flight. Provisions have been made to allow automatic clustering, automatic or manual mainstage initiation, automatic or manual control of heater power, and internal multiplexing of control system instrumentation signals.

SPARK IGNITION SYSTEM

Ignition is established in the gas generator and main chamber with redundant high-tension spark ignition systems. The spark exciters transform 28 vdc into 15,000 v which discharges across the spark gap at a rate of 50 sparks per sec. Transmission cabling and connections are hermetically sealed to ensure proper operation at high altitudes.

POWER REQUIREMENTS

Direct current power is used for the following:

1. Heaters to maintain controller components at an operational temperature while in an environmental temperature as low as -250 F. (A temperature of -200 F is anticipated in the vehicle engine compartment when propellants are tanked.) Heaters are turned off when the engine is started. In the event it is intended to utilize engine restart capabilities, the heaters will re-energize on engine shutdown if a control signal is present from the vehicle.
2. Energizing control sequencing system
3. Propellant ignition systems
4. Control components solenoids

Alternating current must be provided by the vehicle system to power and control the propellant utilization valve system (section 11).

Power definition and usage requirements for both dc and ac current are presented in Fig. 8.1 and as follows:

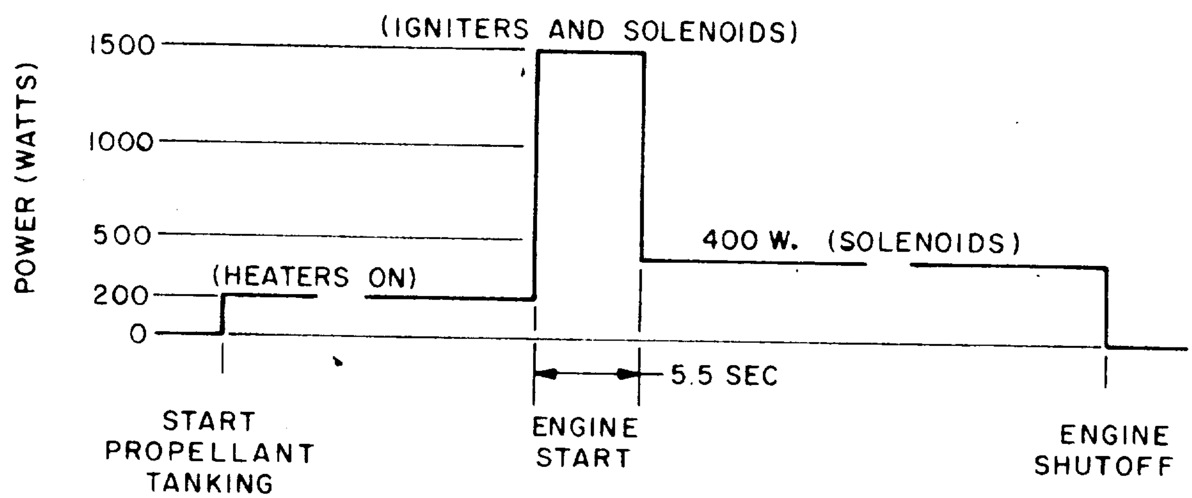
<u>Direct Current</u>	24 to 30 vdc, 32 v max
Continuous	200 w (required for heater power only during long hold periods; heaters may be turned off at engine start)

Start	1500 w for 5.5 sec (spark exciters solenoid control)
Powered Flight	400 w (solenoid control power)
<u>Alternating Current</u>	108 to 121 vac, 400 cps, 100 w continuous for operation of the propellant utilization system

A schematic diagram of the engine electrical system defining the engine and vehicle connections to the engine is shown in Fig. 8.2.

NOTE: The control system is designed for a maximum no damage voltage of 32 volts; satisfactory ignition and system timing will not be obtained if the voltage is below 24 volts.

24 TO 30 VOLT DC POWER USAGE

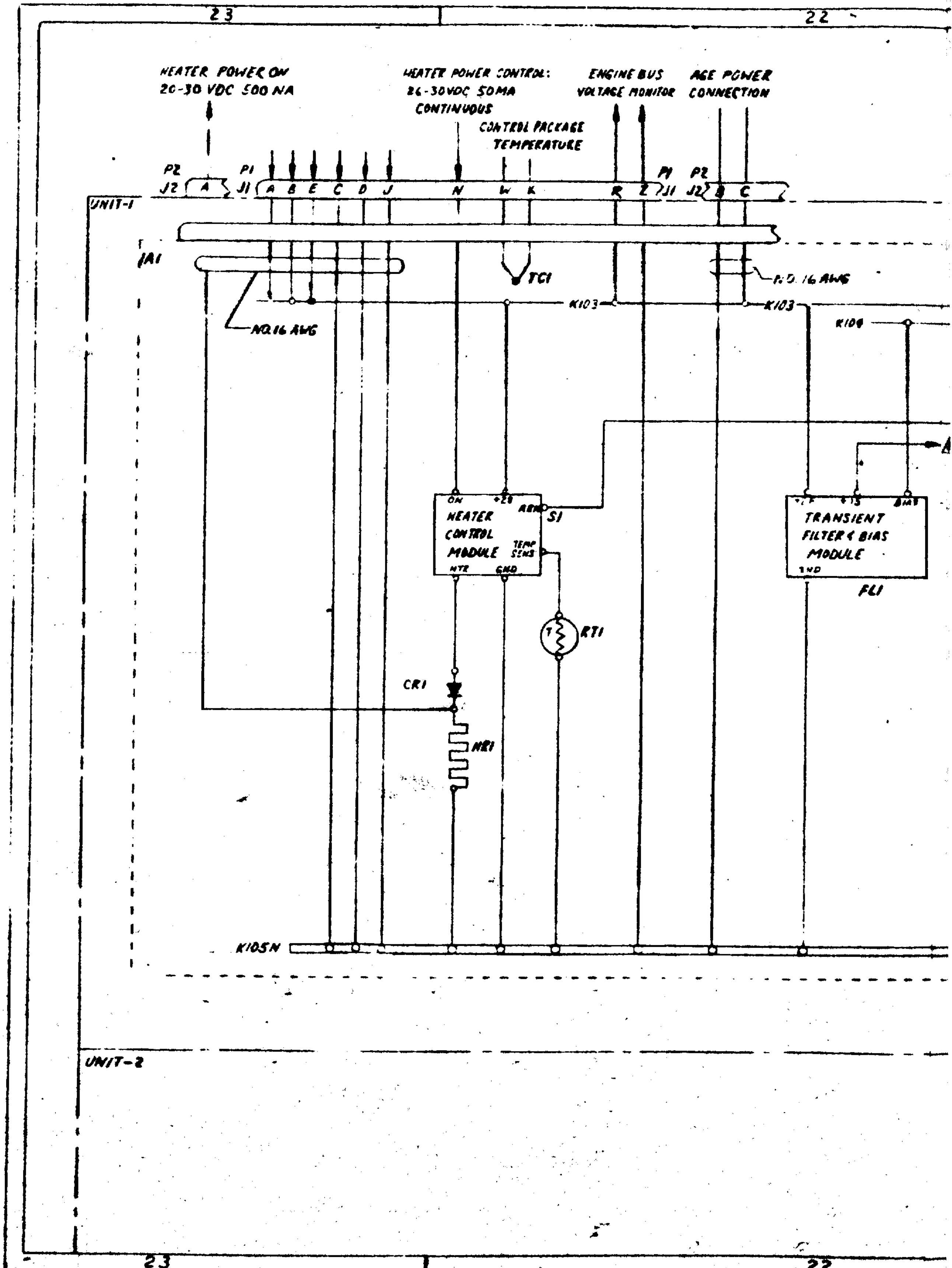


108 TO 121 VAC, 400 CPS, SINGLE PHASE POWER USAGE

CONTINUOUS 100 W USE FROM TIME OF ENGINE START TO SHUTOFF.

Figure 8.1. Engine Electrical Power Requirement Schedule

[redacted]



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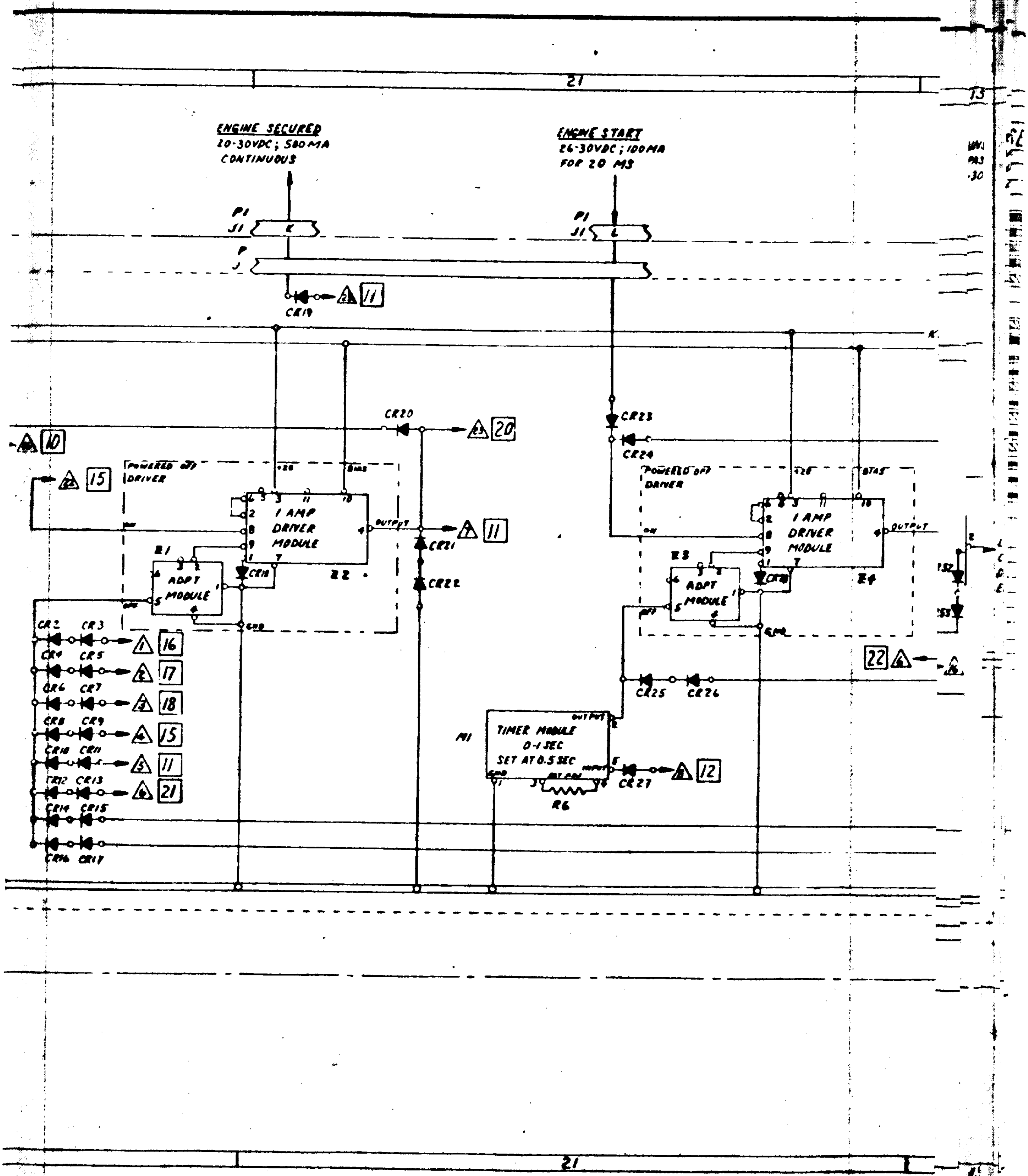
R-2661-4P

FORM R 18-4

DIETERICH-POST CLEARPRINT 1000H



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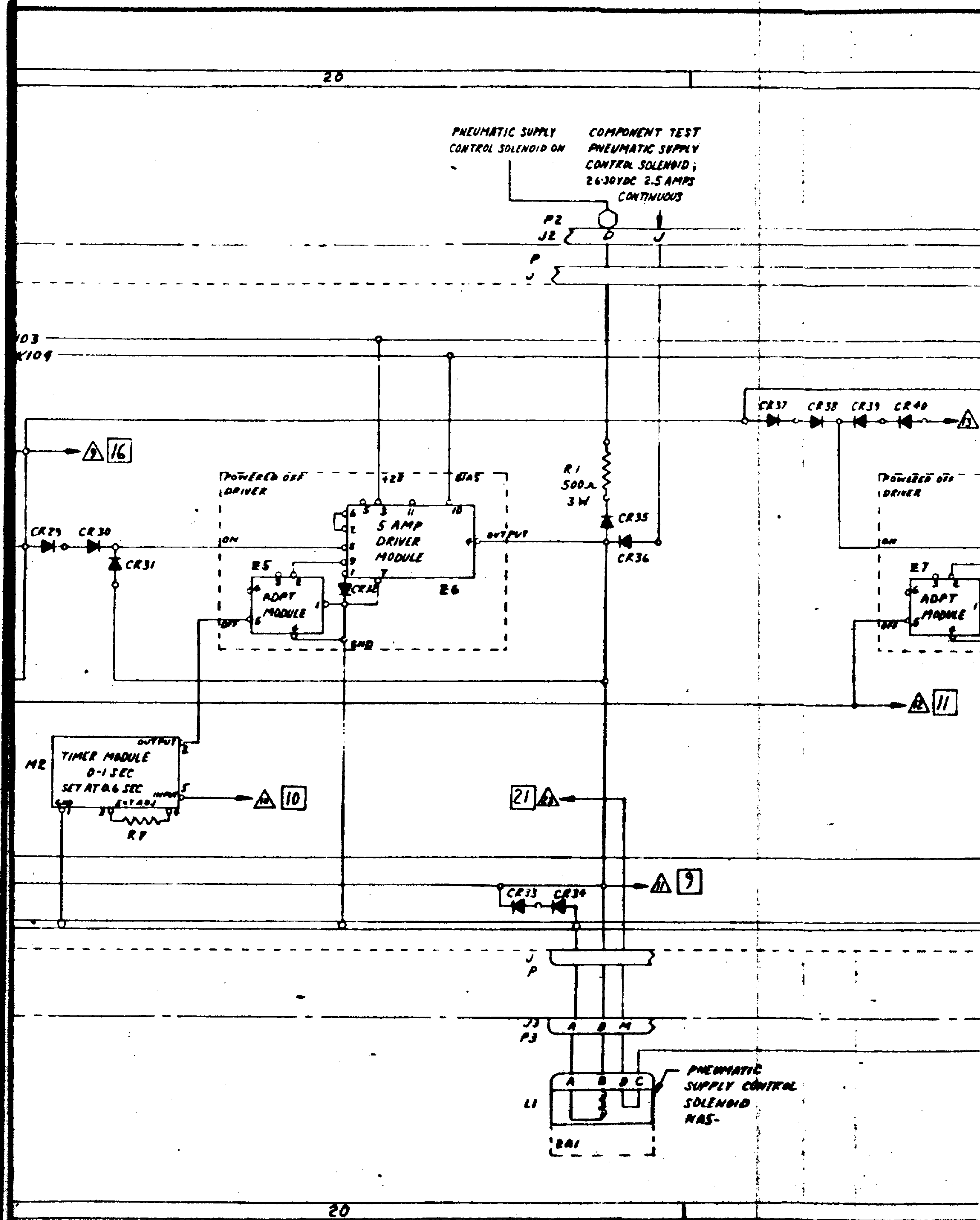

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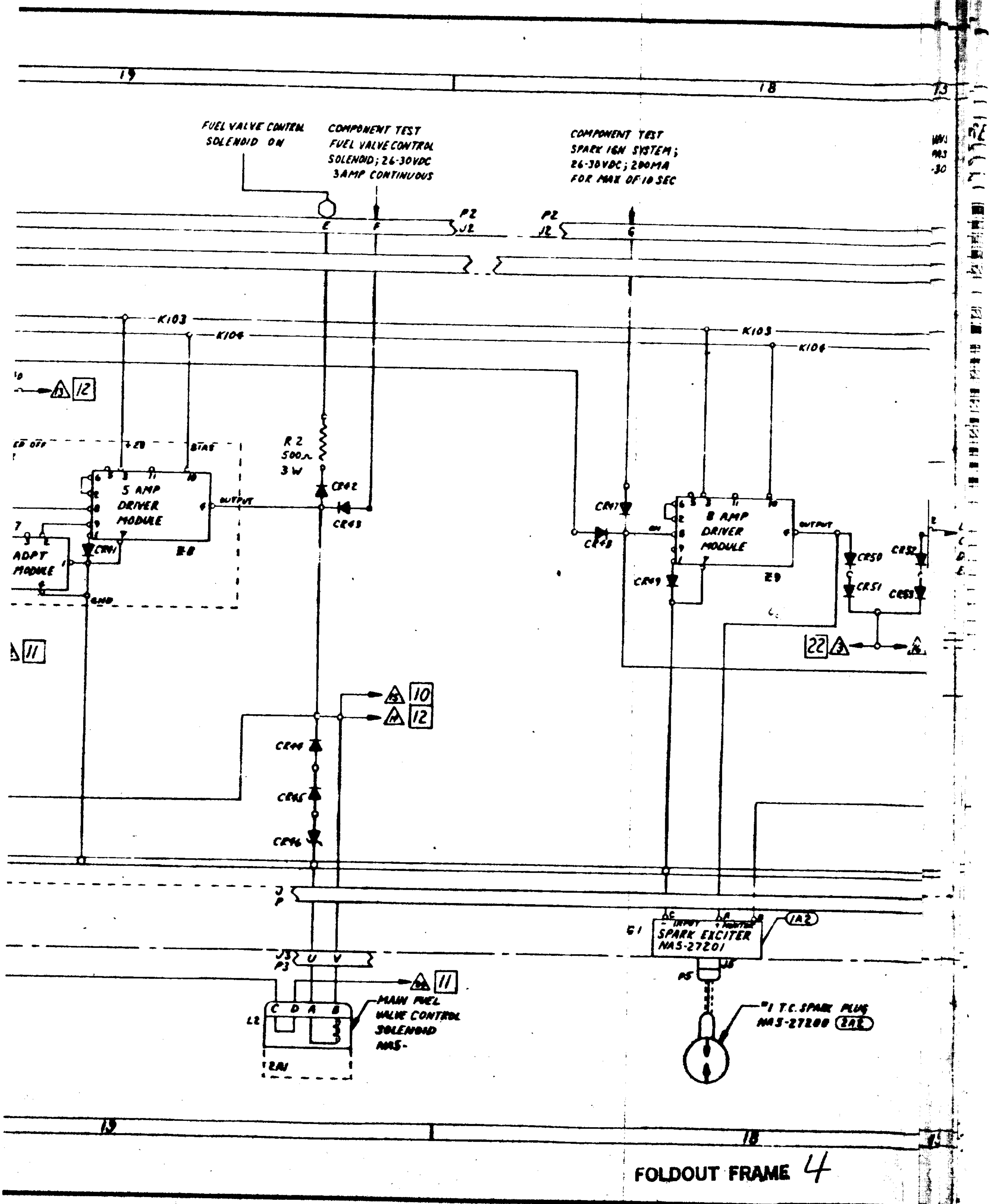

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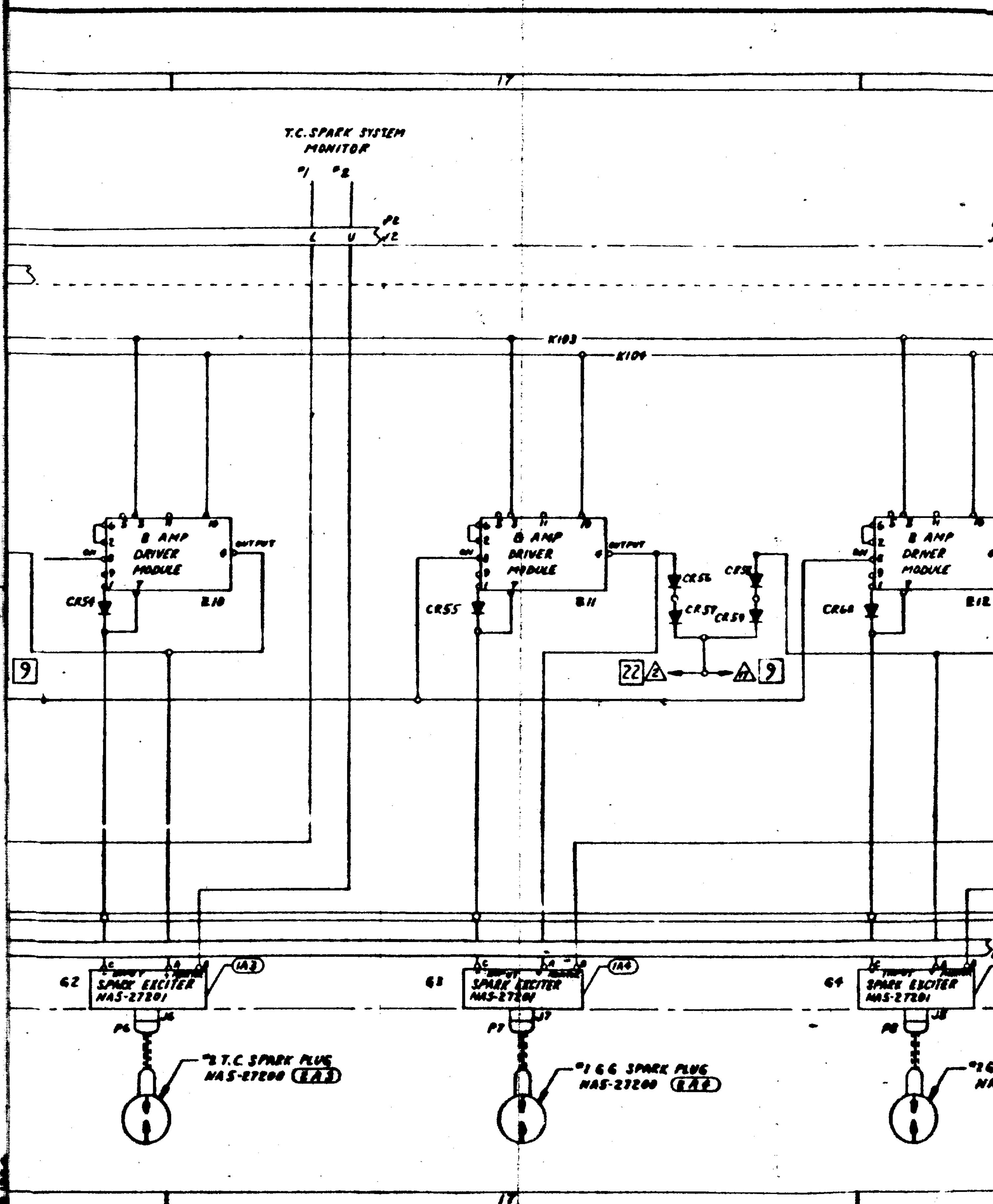

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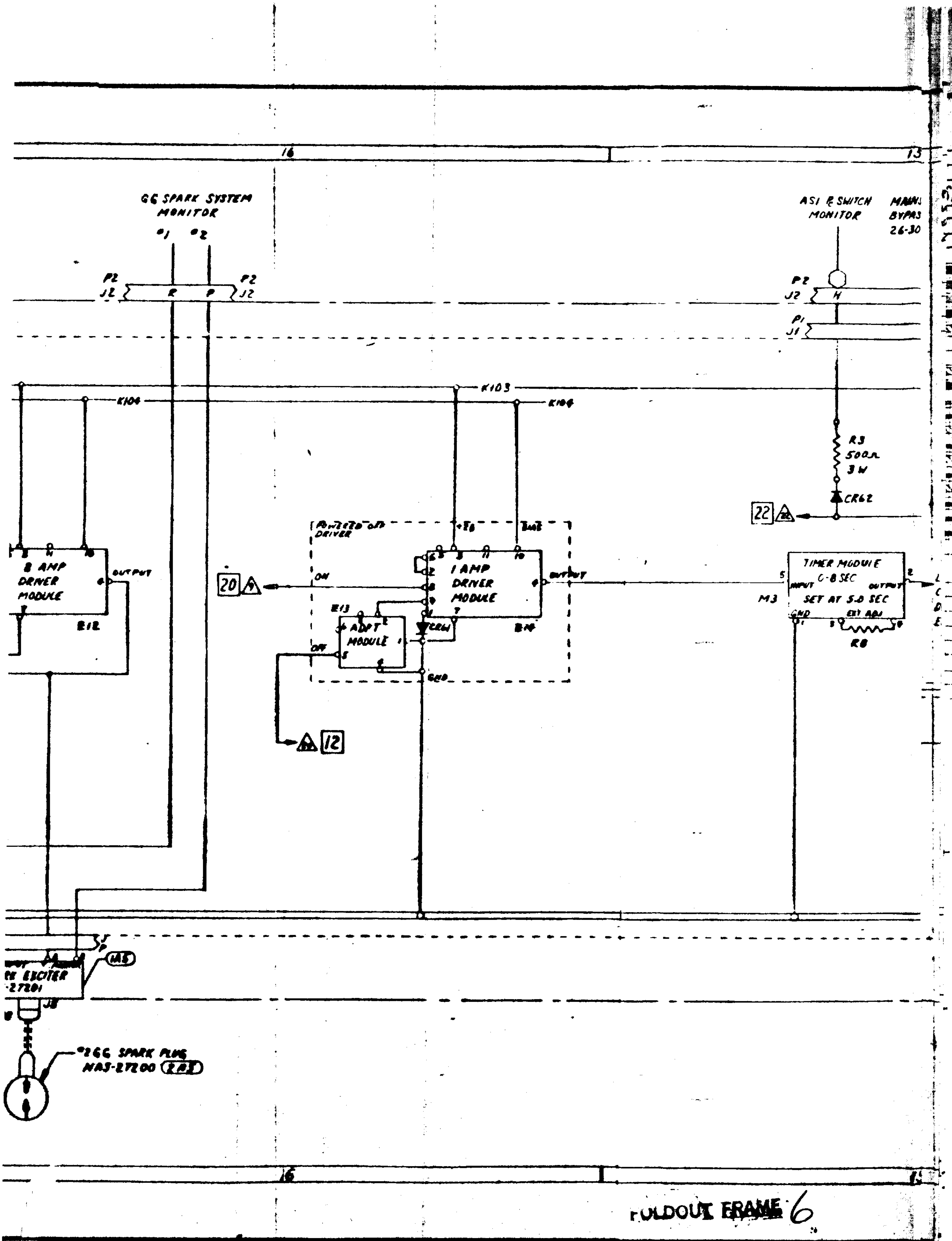

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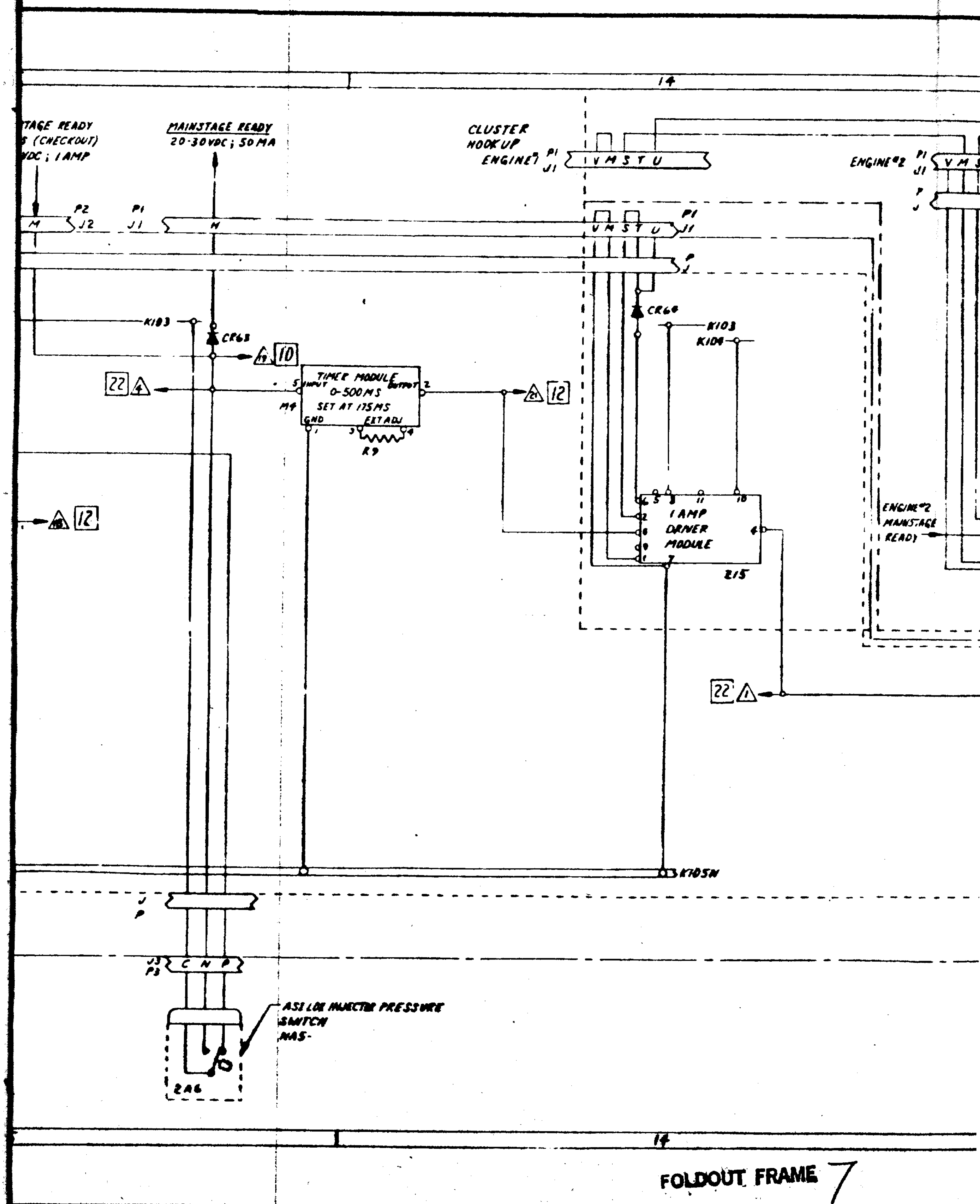
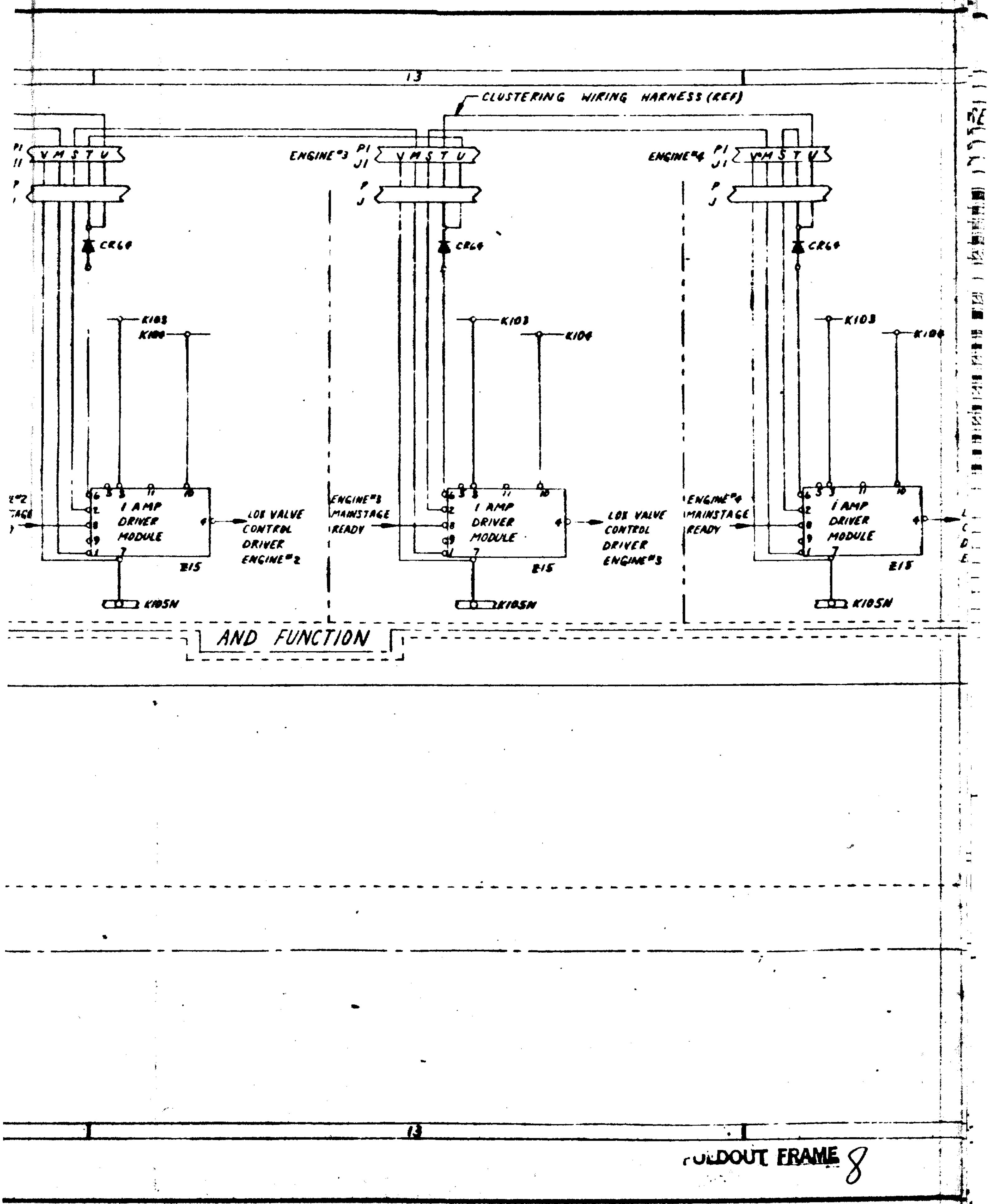
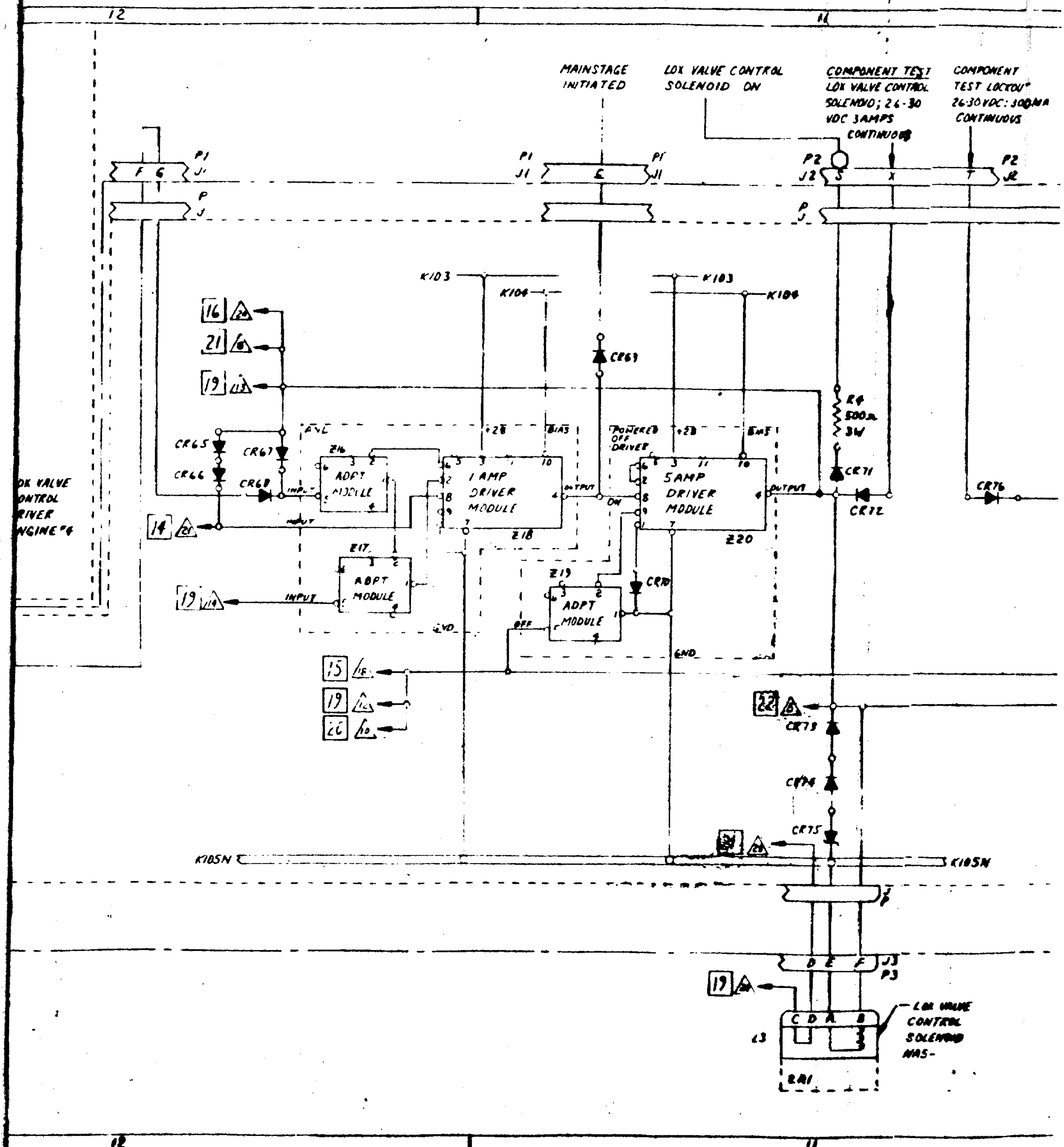
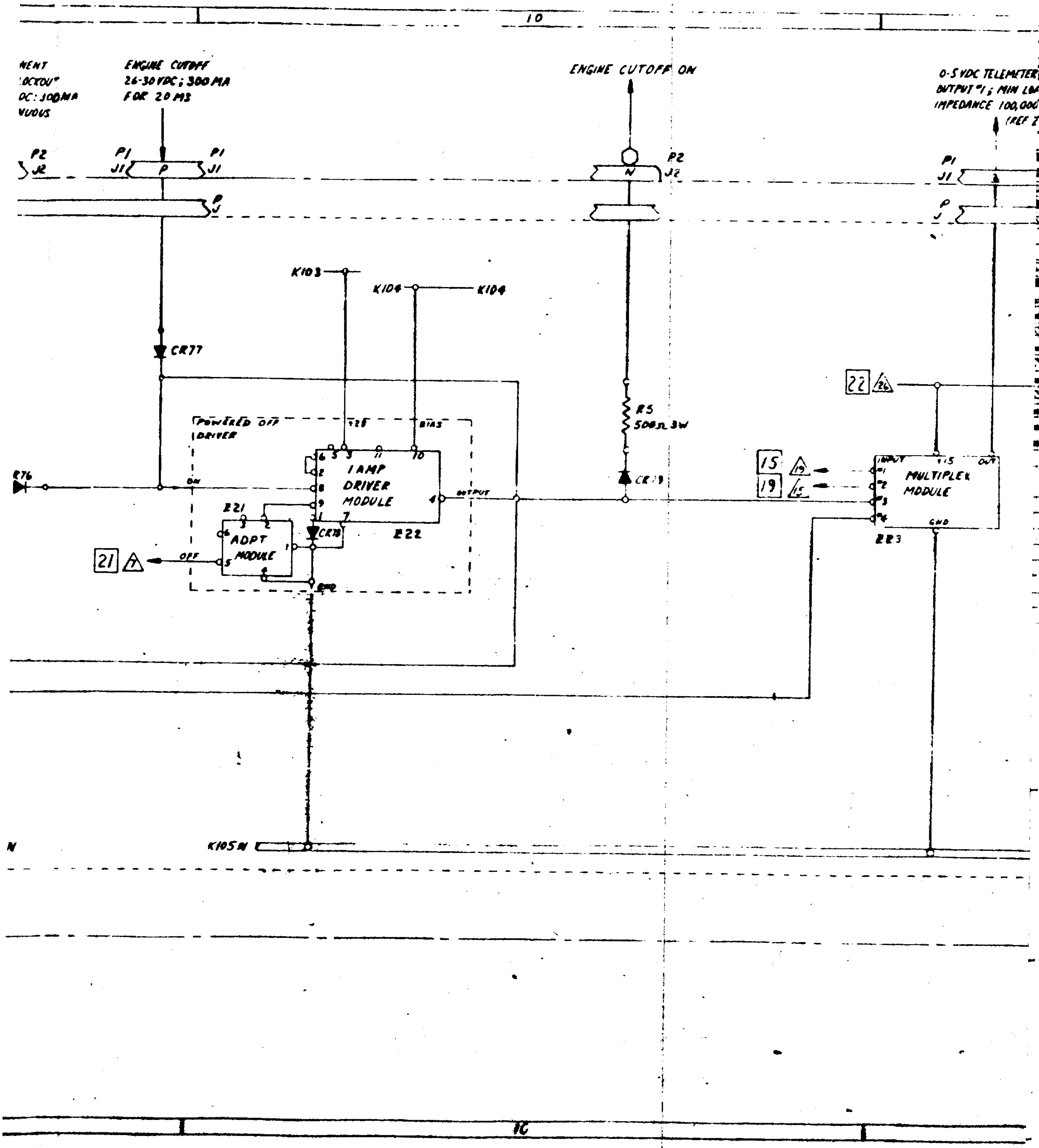

14
MAINSTAGE READY
20-30VDC; 50MA
CLUSTER
HOOK UP
ENGINE
P1
J1
V M S T U
ENGINE #2
P2
J2
M
K103
CR63
CR64
K104
10
12
22
4
TIMER MODULE
0-500MS
SET AT 175MS
INPUT
GND
EXT ADJ
M4
R9
1 AMP
DRIVER
MODULE
Z15
ENGINE #2
MAINSTAGE
READY
K105N
J
P
J3
P3
C N P
ASI LOX INJECTOR PRESSURE
SWITCH
MAS-
2A6
14



FOLDOUT FRAME 8



12
11
MAINSTAGE INITIATED
LOX VALVE CONTROL SOLENOID ON
COMPONENT TEST LOX VALVE CONTROL SOLENOID; 26-30 VDC 3AMPS CONTINUOUS
COMPONENT TEST LOCKOUT 26-30 VDC: 300MA CONTINUOUS
K103
K104
CR69
R4 500Ω 3W
CR71
CR72
CR76
CR65
CR66
CR67
CR68
Z16 ADPT MODULE
1 AMP DRIVER MODULE
Z18
5AMP DRIVER MODULE
Z20
Z17 ADPT MODULE
Z19 ADPT MODULE
CR70
POWERED OFF DRIVER
OUTPUT
INPUT
ON
OFF
GND
BIAS
+28
CR73
CR74
CR75
K105N
LOX VALVE CONTROL SOLENOID



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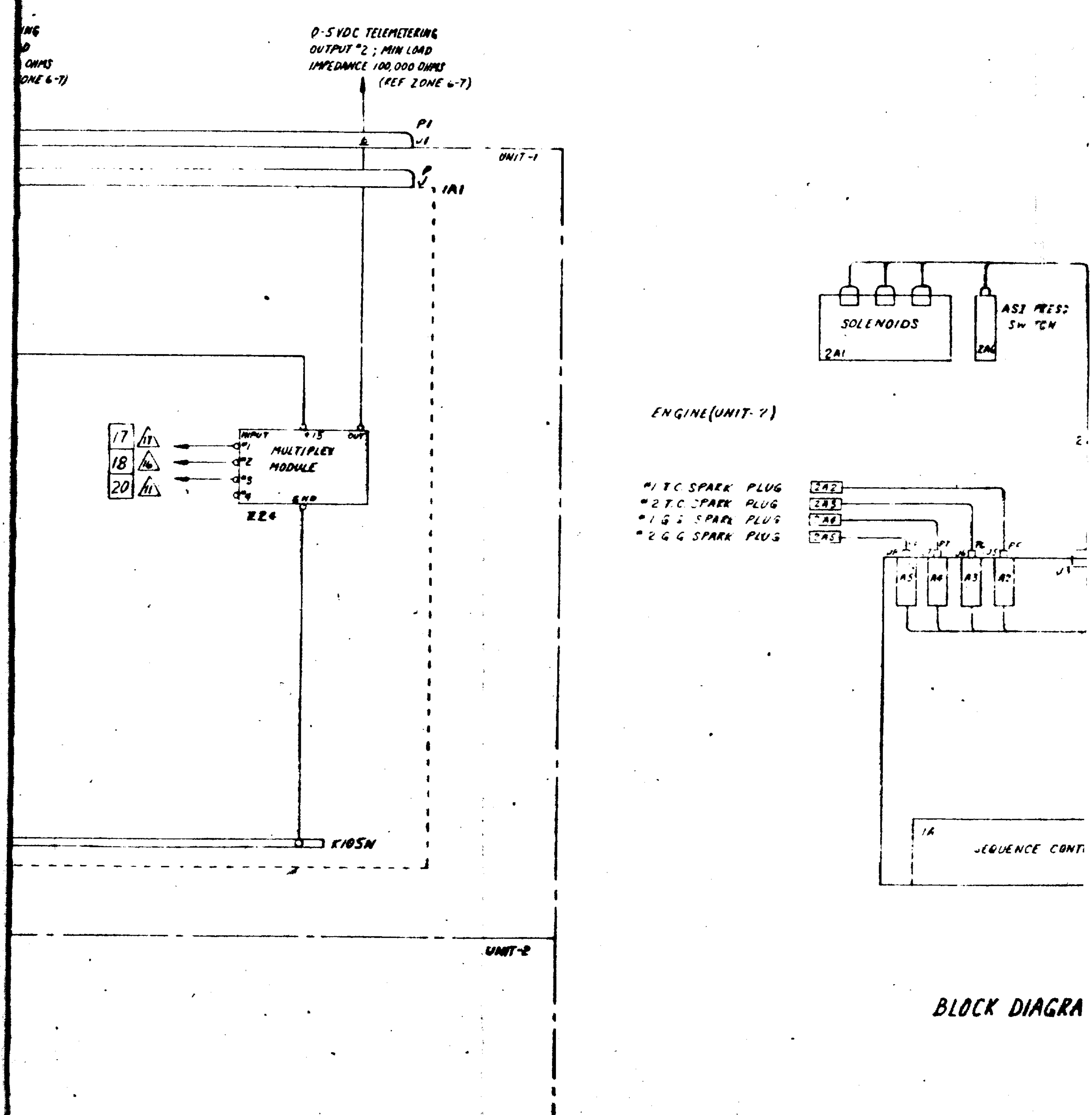
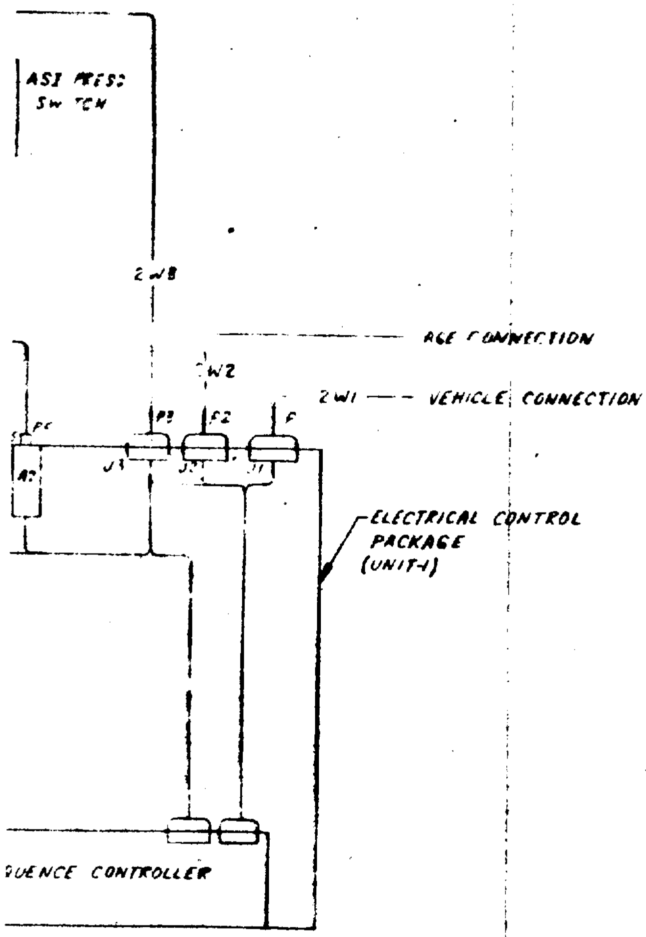


BLOCK DIAGRA

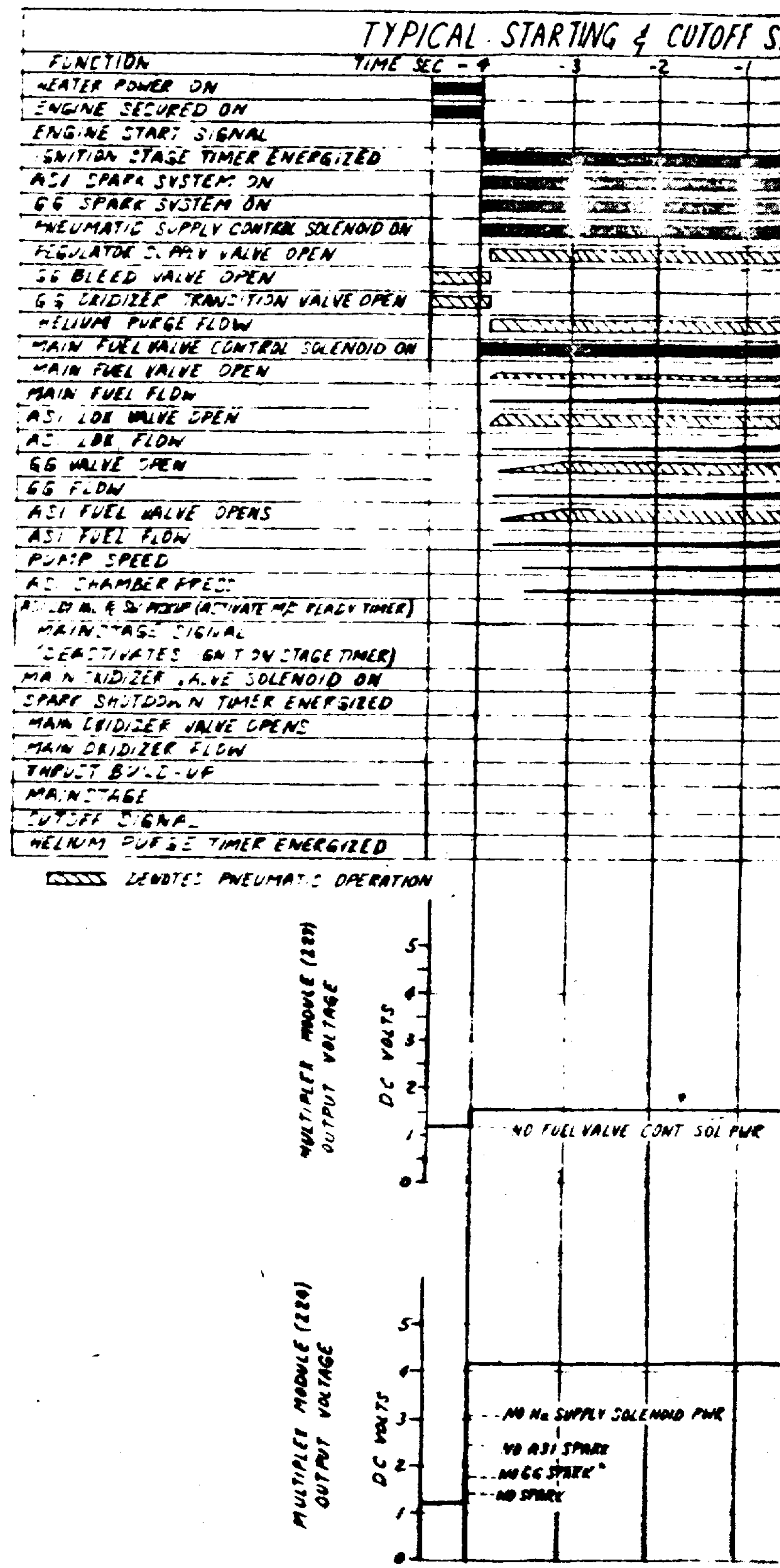
9

8

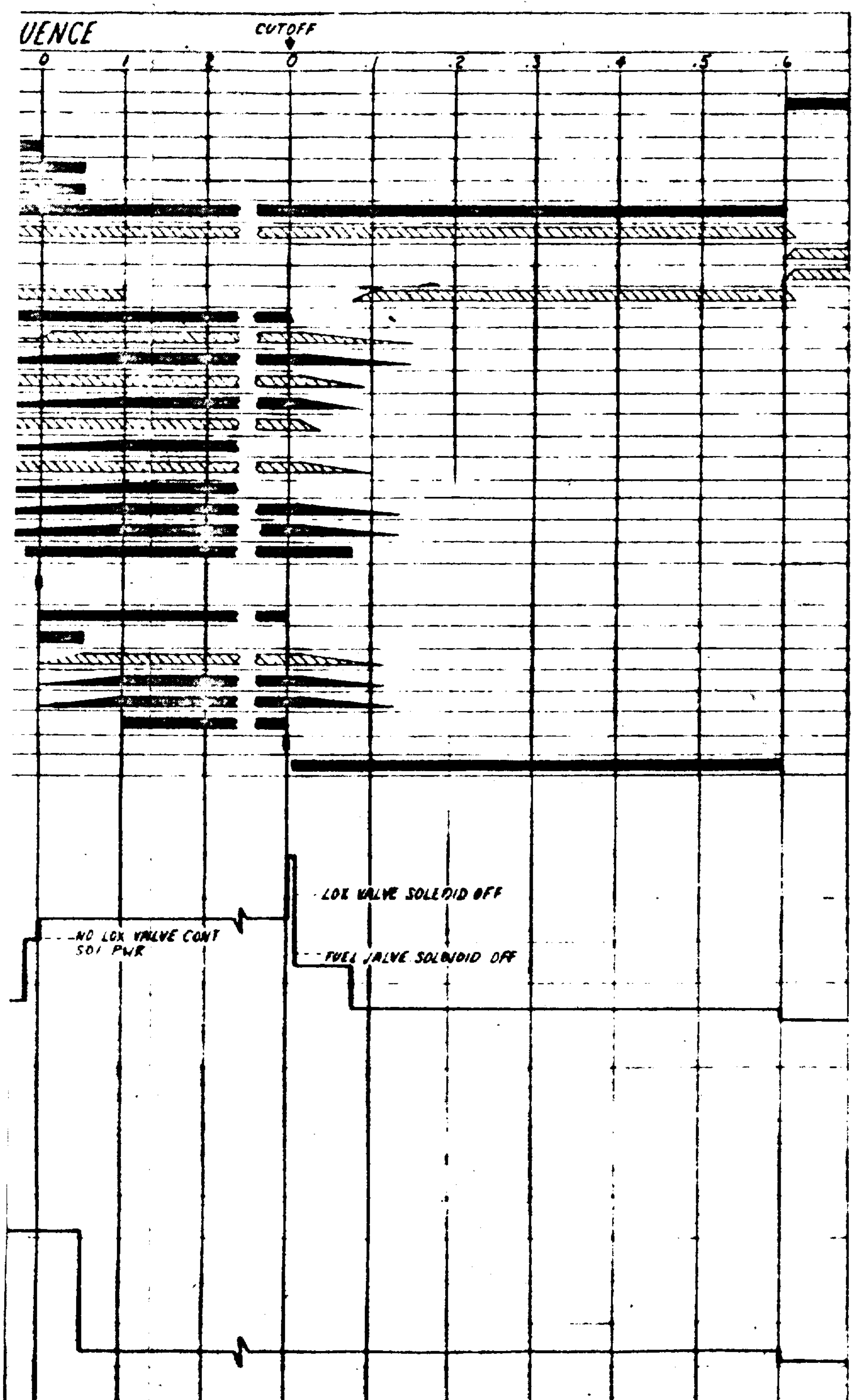
FOLDOUT FRAME //


ASI PRESS SWITCH
2W8
AGE CONNECTION
2W2
2W1 — VEHICLE CONNECTION
P3
P2
P1
J3
J2
J1
ELECTRICAL CONTROL PACKAGE (UNIT-1)
SEQUENCE CONTROLLER

Y DIAGRAM


TYPICAL STARTING & CUTOFF SE
FUNCTION
TIME SEC - 4
-3
-2
-1
HEATER POWER ON
ENGINE SECURED ON
ENGINE START SIGNAL
IGNITION STAGE TIMER ENERGIZED
ASI SPARK SYSTEM ON
GG SPARK SYSTEM ON
PNEUMATIC SUPPLY CONTROL SOLENOID ON
REGULATOR SUPPLY VALVE OPEN
GG BLEED VALVE OPEN
GG OXIDIZER TRANSITION VALVE OPEN
HELIUM PURGE FLOW
MAIN FUEL VALVE CONTROL SOLENOID ON
MAIN FUEL VALVE OPEN
MAIN FUEL FLOW
ASI LOX VALVE OPEN
ASI LOX FLOW
GG VALVE OPEN
GG FLOW
ASI FUEL VALVE OPENS
ASI FUEL FLOW
PUMP SPEED
ASI CHAMBER PRESS
MAINSTAGE SIGNAL
(DEACTIVATES IGNITION STAGE TIMER)
MAIN OXIDIZER VALVE SOLENOID ON
SPARK SHUTDOWN TIMER ENERGIZED
MAIN OXIDIZER VALVE OPENS
MAIN OXIDIZER FLOW
THRUST BUILD-UP
MAINSTAGE
CUTOFF SIGNAL
HELIUM PURGE TIMER ENERGIZED
DENOTES PNEUMATIC OPERATION
MULTIPLEX MODULE (2B1) OUTPUT VOLTAGE
DC VOLTS
NO FUEL VALVE CONT SOL PWR
MULTIPLEX MODULE (2B2) OUTPUT VOLTAGE
DC VOLTS
NO He SUPPLY SOLENOID PWR
NO ASI SPARK
NO GG SPARK
NO SPARK

6 | 5



CONNECTORS

ITEM NO.	PART NO.	PINS	ZONE
1J1			9-23
1P1			9-23
1J2			10-23
1P2			10-23
1J3			11-20
1P3			11-20
1A1J1			9-23
1A1P1			9-23
1A1J2			10-20
A1P2			10-20

RESISTORS

ITEM NO.	OHMS	WATTS	PART NO	ZONE
1A1R1	500	3		20
1A1R2	500	3		19
1A1R3	500	3		15
1A1R4	500	3		11
1A1R5	500	3		10
A1R6				21
1A1R7				20
1A1R8				15
1A1R9				14

M

ITEM NO.	PAR
1A1Z1	
1A1Z2	
1A1Z3	
1A1Z4	
1A1Z5	
1A1Z6	
1A1Z7	
1A1Z8	
1A1Z9	
A1Z10	
1A1Z11	
1A1Z12	
1A1Z13	
1A1Z14	
1A1Z15	
1A1Z16	
1A1Z17	
1A1Z18	
1A1Z19	
1A1Z20	
1A1Z21	
1A1Z22	
1A1Z23	
1A1Z24	

TIM

ITEM NO.	PART
1A1M1	
A1M2	
A1M3	
1A1M4	

MISC

ITEM NO.	PART
1A1FL1	
1A1HR1	
1A1RT1	
1A1S1	
1A1TC1	

6 | 5

5	4

CONNECTORS (partial)

PART NO.	PINS	ZONE
		9-23
		9-23
		10-23
		10-23
		11-20
		11-20
		9-23
		9-23
		10-20
		10-20

MODULES

ITEM NO.	PART NO.	DESCRIPTION	ZONE
1A1Z1		ADPT	22
1A1Z2		1 AMP	21
1A1Z3		ADPT	21
1A1Z4		1 AMP	21
1A1Z5		ADPT	20
1A1Z6		5 AMP	20
1A1Z7		ADPT	19
1A1Z8		5 AMP	19
1A1Z9		8 AMP	18
1A1Z10		8 AMP	17-18
1A1Z11		8 AMP	17
1A1Z12		8 AMP	16
1A1Z13		ADPT	16
1A1Z14		1 AMP	16
1A1Z15		1 AMP	14
1A1Z16		ADPT	12
1A1Z17		ADPT	12
1A1Z18		1 AMP	11-12
1A1Z19		ADPT	11
1A1Z20		5 AMP	11
1A1Z21		ADPT	10
1A1Z22		1 AMP	10
1A1Z23		MULTIPLEX	9-10
1A1Z24		MULTIPLEX	9

TIMERS

ITEM NO.	PART NO.	DESCRIPTION	ZONE
1A1M1		0-1 SEC	21
1A1M2		0-1 SEC	20
1A1M3		0-8 SEC	15
1A1M4		0-500 MS	14-15

RESISTORS (partial)

WATTS	PART NO	ZONE
3		20
3		19
3		15
3		11
3		10
		21
		20
		15
		14

MISC

ITEM NO.	PART NO.	DESCRIPTION	ZONE
1A1FL1			22
1A1HR1			22
1A1RT1			22
1A1S1			22-23
1A1TC1			22

RECTIFIERS

ITEM NO.	PART NO.	ZONE
1A1CR1		23
CR2		22
CR3		
CR4		
CR5		
CR6		
CR7		
CR8		
CR9		
CR10		
CR11		
CR12		
CR13		
CR14		
CR15		
CR16		
CR17		22
CR18		21
CR19		
CR20		
CR21		
CR22		
CR23		
CR24		
CR25		
CR26		
CR27		
CR28		21
CR29		20
CR30		
CR31		
CR32		
CR33		
CR34		
CR35		
CR36		20
CR37		19
CR38		
CR39		
CR40		
CR41		
CR42		
CR43		
CR44		
CR45		
CR46		19
CR47		18
CR48		
CR49		
CR50		
CR51		
CR52		
CR53		
CR54		18
CR55		17
CR56		
CR57		
CR58		
CR59		
CR60		17
CR61		16
CR62		15
1A1CR63		15

RECTIFIERS (CONT)

ITEM NO.	PART NO
1A1CR64	
CR65	
CR66	
CR67	
CR68	
CR69	
CR70	
CR71	
CR72	
CR73	
CR74	
CR75	
CR76	
CR77	
CR78	
1A1CR79	

SYMBOLS NOT S... MIL-STD-15

⬡	DENOTES INSTRUMENTAT... MIN LOAD IMPEDANCE ... NON INDUCTIVE
(zener symbol)	ZENER DIODE
□	ZONE REF BLOCK
△	CORRESPONDING DELTA... NUMBERS INDICATE CI... CONTINUATION - LAST...
(heater symbol)	HEATER
(pressure switch symbol)	PRESSURE S...

5	4

ZONE
14
12
12
12
12
11
11
10
13

SUB ASSYS OF UNIT 1

ITEM NO.	PART NO.	DESCRIPTION	ZONE
1A1		SEQUENCE CONTROLLER	9-23
1A2		SPARK EXCITER	18
1A3		SPARK EXCITER	18
1A4		SPARK EXCITER	17
1A5		SPARK EXCITER	16

SUB ASSYS OF UNIT 2

ITEM NO.	PART NO.	DESCRIPTION	ZONE
2A1		PNEUMATIC PACKAGE	4,20(B)
2A2		#1 T.C. SPARK PLUG	18
2A3		#2 T.C. SPARK PLUG	17
2A4		#1 G.G. SPARK PLUG	17
2A5		#2 G.G. SPARK PLUG	16
2A6		ASI LOX INJ PRESS SW	15

HOWN IN

ON SIGNAL
2 K OHM

RCUIT
NG USED 26

ITCH

FOLDOUT FRAME 15

Figure 8.2. Electrical System Schematic

[redacted]

INSTRUMENTATION

ENGINE SYSTEM INSTRUMENTATION

Instrumentation provided on the J-2 engine consists of propellant pump discharge line flowmeters, pump speed signals, and various electrical sequence signals.

Signal pick-up for telemetering or other vehicle use is through a common multiple pin connector used only for instrumentation connection and is provided from the electrical pneumatic package. These are the only instruments to be delivered as part of the engine. However, instrumentation tap provisions also are made in anticipation of the users needs for flight.

FLIGHT INSTRUMENTATION PROVISIONS

A list of recommended flight instrumentation is shown in Table 9.1

Flowmeters

The basic element of the flowmeter is a helical-vaned rotor which is turned by propellant flow to measure flow velocity. The rotor is free-turning on an anti-friction bearing system. The flow diameter is closely controlled to permit accurate determination of the volumetric flowrate. The hydrogen meter has a four-vane rotor producing four electrical impulses per revolution and turns approximately 3600 rpm at nominal flow.

TABLE 9.1

RECOMMENDED J-2 FLIGHT INSTRUMENTATION

Parameter	Range
Electrical Control System Signals	
Engine Bus Voltage	0 to 32 vdc
Control Box Temperature	-300 to 200 F
Liquid Oxygen Pump Speed	0 to 11,000 rpm
Hydrogen Pump Speed	0 to 30,000 rpm
ASI Pressure Switch Pickup	*
Helium Supply Solenoid Energized	*
ASI Spark Ignition System Energized	*
GG Spark Ignition System Energized	*
Mainstage Ready Signal	*
Oxygen Control Solenoid Energized	*
Cutoff Signal	*
Fuel Control Solenoid Energized	*
Primary Flight Instrumentation Signals	
Thrust Chamber Pressure	0 to 800 psia
Helium Supply Pressure	0 to 5000 psia
Gas Generator Pressure	0 to 800 psia
Propellant Utilization Valve Position	0 to 2000 ohm
Hydrogen Turbine Inlet Temperature	0 to 2000 F
Helium Supply Temperature	-400 to 100 F

*Reference zone 6 and 9 on Fig. 8.2 electrical system schematic

TABLE 9.1

(Continued)

Parameter	Range
Liquid Oxygen High-Pressure Duct Pressure	0 to 1300 psia
Liquid Oxygen High-Pressure Duct Temperature	-300 ±20 F
Liquid Oxygen Flow	
Hydrogen High-Pressure Duct Pressure	0 to 1300 psia
Hydrogen High-Pressure Duct Temperature	-425 ±10 F
Hydrogen Flow	
Gas Generator Liquid Oxygen Injection Pressure	0 to 100 psia
Gas Generator Fuel Injection Pressure	0 to 1000 psia
Main Liquid Oxygen Injection Pressure	0 to 1000 psia
Main Fuel Injection Pressure	0 to 1000 psia
Liquid Oxygen Turbine Inlet Pressure	0 to 150 psia
Engine Regulator Outlet Pressure	0 to 500 psia
Balanced Piston Sump Pressure (Hydrogen Pump)	0 to 500 psia
Main Liquid Oxygen Valve Position	0 to 2000 ohm
Main Hydrogen Valve Position	0 to 2000 ohm
Hydrogen Temperature at Thrust Chamber Tapoff	-450 to 50 F
Gas Generator Valve Position	0 to 2000 ohm
Fuel Pump Inlet Pressure	0 to 100 psia
Fuel Pump Inlet Temperature	-425 ±10 F
Oxidizer Pump Inlet Pressure	0 to 150 psia
Oxidizer Pump Inlet Temperature	-300 ±20 F

The oxidizer meter has a six-vane rotor producing six electrical impulses per revolution and turns approximately 2400 rpm at nominal flow. These signals are made available to the vehicle telemetering system as a 1 volt peak-to-peak signal for the oxidizer flowmeter and a 2 volt peak-to-peak signal for the fuel flowmeter.

PUMP INLET PRESSURE MEASUREMENT

To fully evaluate the propulsion system during flight, the vehicle manufacturer must supply instrumentation for measuring the oxidizer and fuel pump inlet pressures and temperatures. These parameters are not included on the engine because of space limitations. It is therefore suggested that the vehicle manufacturer locate provisions for them in the vehicle duct as close as practical to the engine flange.

TURBOPUMP SPEED INDICATORS

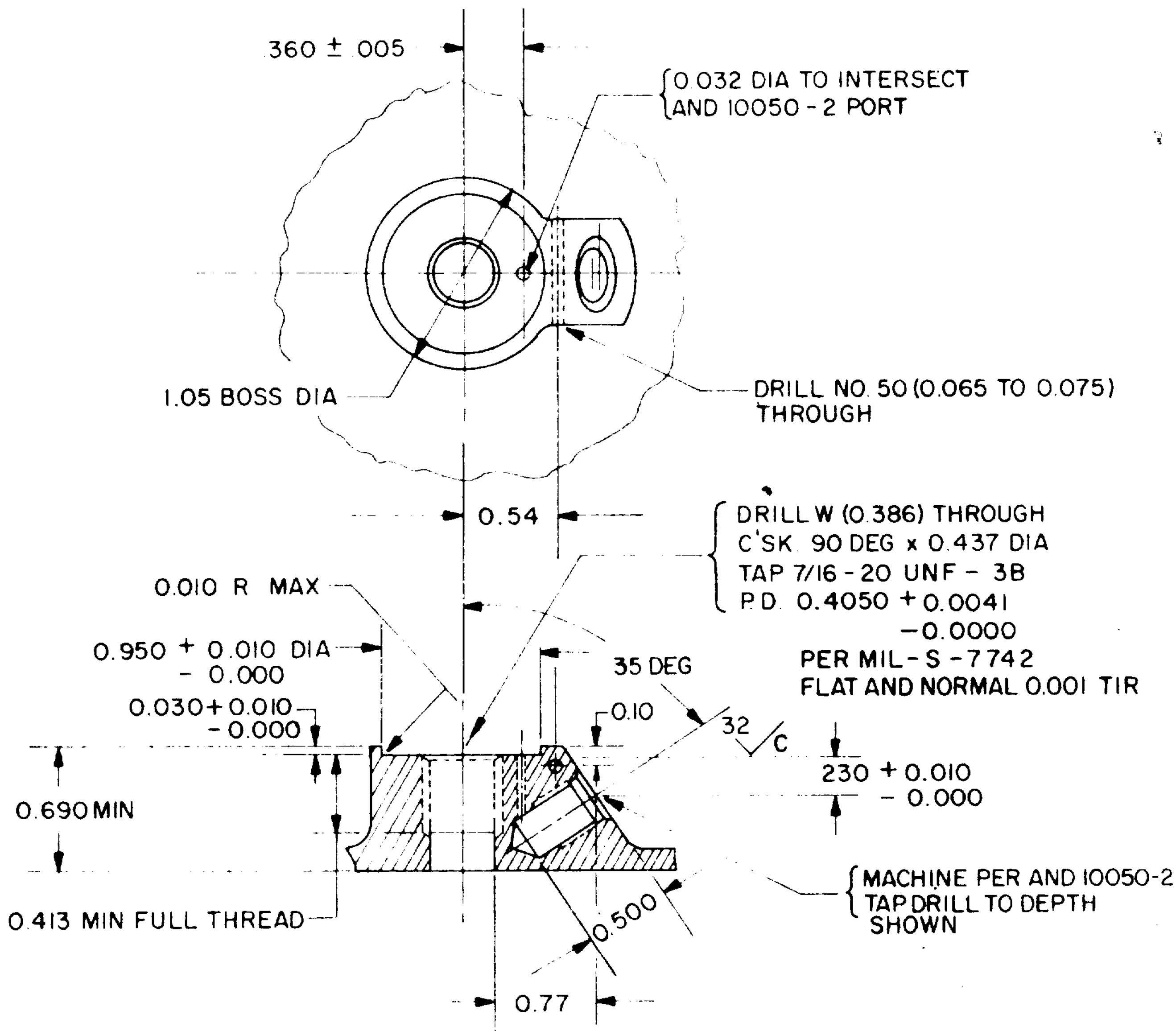
Turbopump speed for both turbopumps is indicated by the output of an a-c generator. The current frequency varies directly with pump speed and is based on 12 cycles per pump revolution. The generator voltage of 100 mv is amplified to 0.5 v and used in the engine sequence control system as well as in the turbopump overspeed shutoff system. The signal is made available to the vehicle telemetering system as a 0.5 vac (variable frequency current).

INSTRUMENTATION PRESSURE PORT PLUGS

Engines will be delivered with ports of a configuration as shown on Fig. 9.1 and 9.2. They will be sealed with a Rocketdyne Naflex seal, P/N 404659, for nominal and low temperature applications, with a Naflex P/N 404661 for temperatures above 160 F. It is recommended that these seals be replaced with a new seal of the same type each time the plug is removed.

A plug as shown in Fig. 9.3 is provided with a vent hole which aligns with the intermediate vent between two concentric seal surfaces on the Naflex seal. This type plug will be provided with ports as shown in Fig. 9.2. All other ports will be provided with blank plugs. An AND 10050-2 port is provided in the plug head connecting to the seal vent. This port is plugged with an MS9015-2 plug, or a dimensional equivalent plug and an MS-29513 O-ring seal or a Rocketdyne P/N RD 261-6001-0001 solid copper gasket. Seal selection is based on fitting temperature.

The AND 10050-2 plug will permit the engine user to check for primary seal leakage under the special plug by connecting a 1/8-in. tube to the port and routing it to a measuring or indicating device.

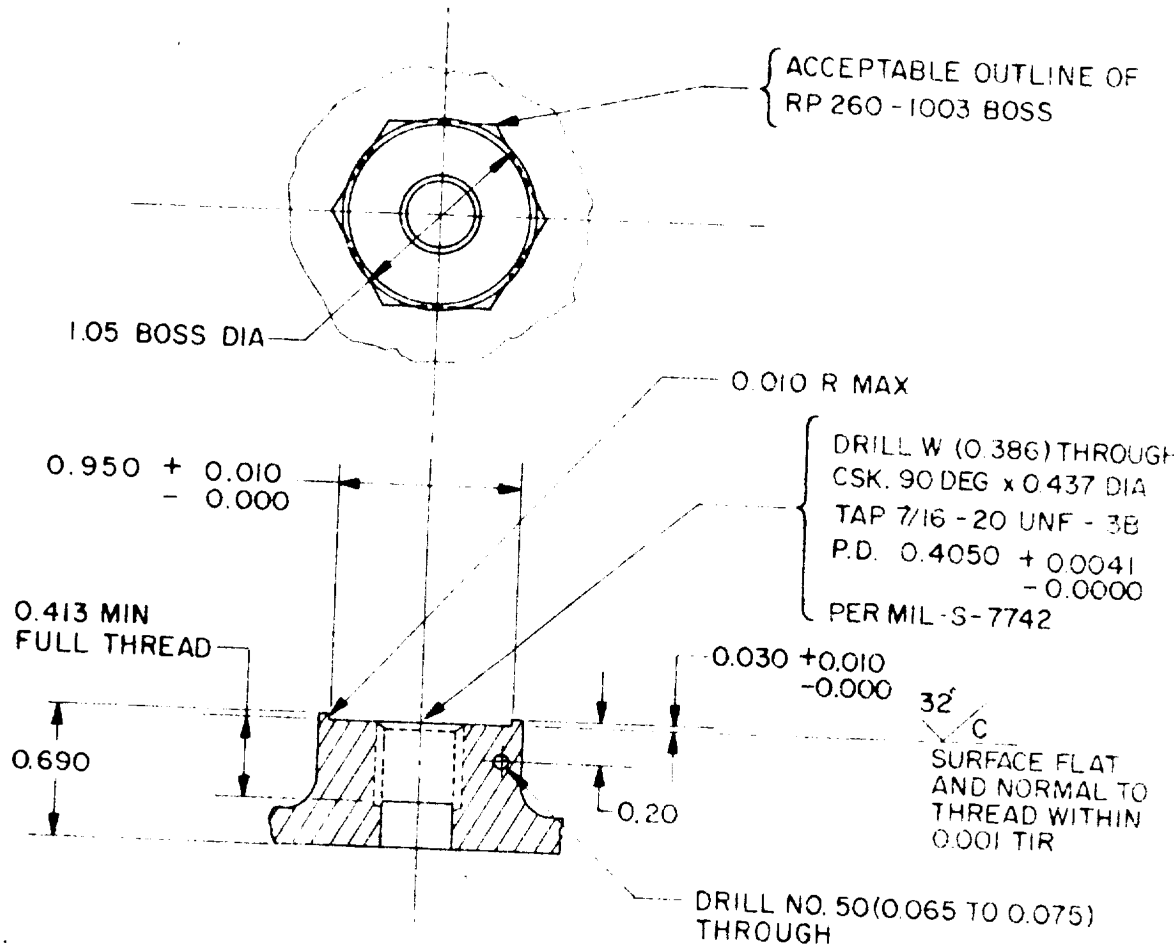


NOTE: THIS DESIGN TO BE USED WITH FITTINGS AT LOCATIONS WHERE IN FLIGHT INSTRUMENTATION IS REQUIRED.

BOSS OUTLINE SHOWN REPRESENTS A TYPICAL PART. ACTUAL OUTLINE TO BE DETERMINED BY COMPONENT DESIGNER.

Figure 9.1. Double-Seal Vented Instrumentation Boss

FOR APPLICATION AS A WELDED BOSS TO COMPONENT OR TUBING WALLS USE RP 260-1003 CONFIGURATION EXCEPT AS OTHERWISE DEMENSIONED OR INDICATED HERE.



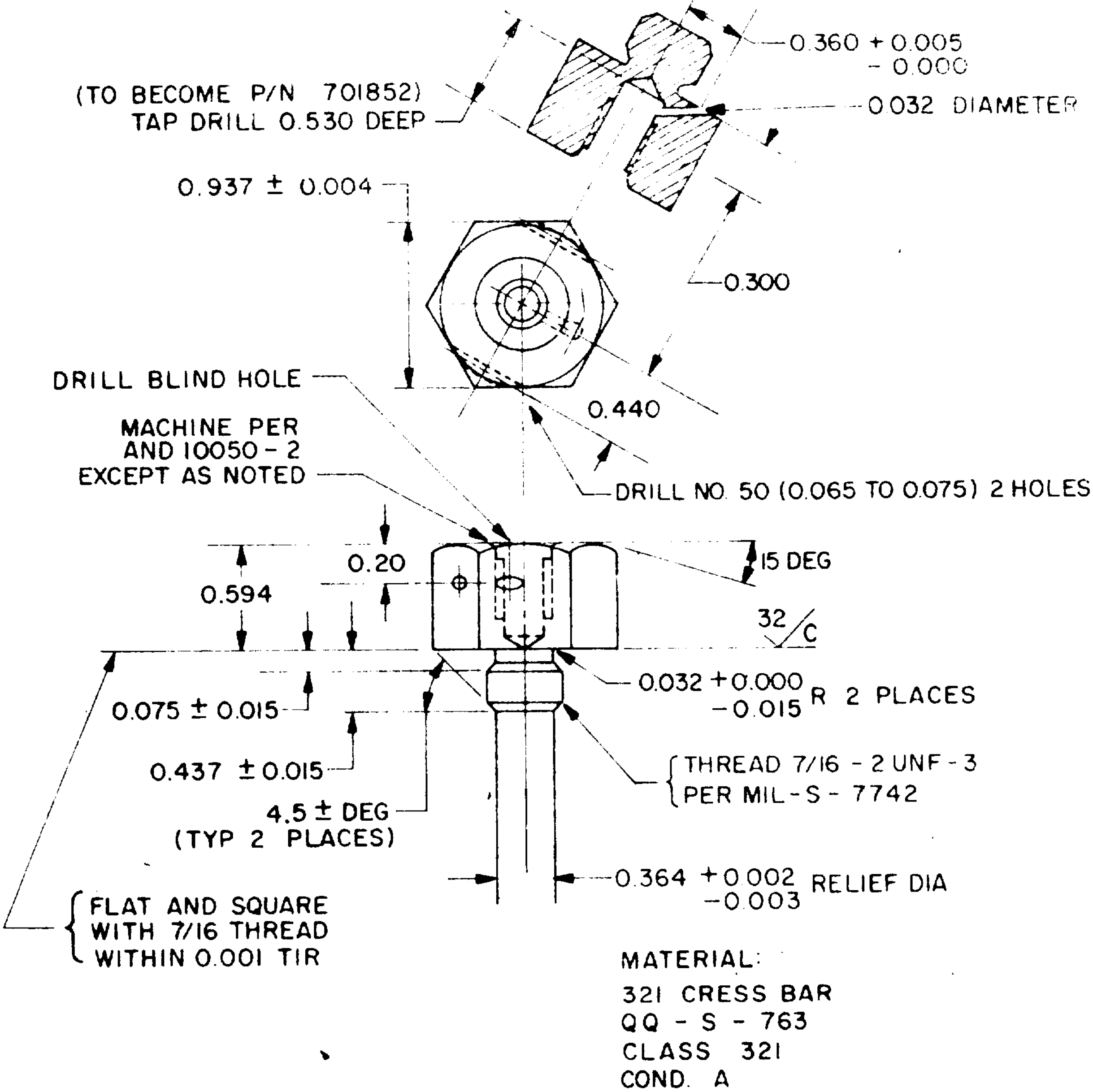
NOTE:

THIS DESIGN TO BE USED IN LOCATIONS WHERE CUSTOMER INSTRUMENTATION IS NOT EXPECTED TO BE REQUIRED.

FOR USE WITH: STATIC PRESSURE PROBES (LOW FREQUENCY), THERMOCOUPLES, AND RESISTANCE TEMPERATURE SENSOR (ROSEMONT)

PLUG TO BE INSTALLED WHEN INSTRUMENTATION IS REMOVED

Figure 9.2. Double-Seal Instrumentation Boss



PLUG USED TO SEAL J2 ENGINE INSTRUMENTATION BOSSES WHICH ARE NOT TO BE USED AFTER ENGINE DELIVERY

AND 10050-2 BOSS IS PROVIDED TO VENT-OFF LEAKAGE OF CRYOGENIC PROPELLANTS THROUGH THE BASIC NAFLEX SEAL P/N 404659 USED WITH PLUG

Figure 9.3. Instrumentation Port Plug

ACCESSORY PROVISIONS

ACCESSORIES

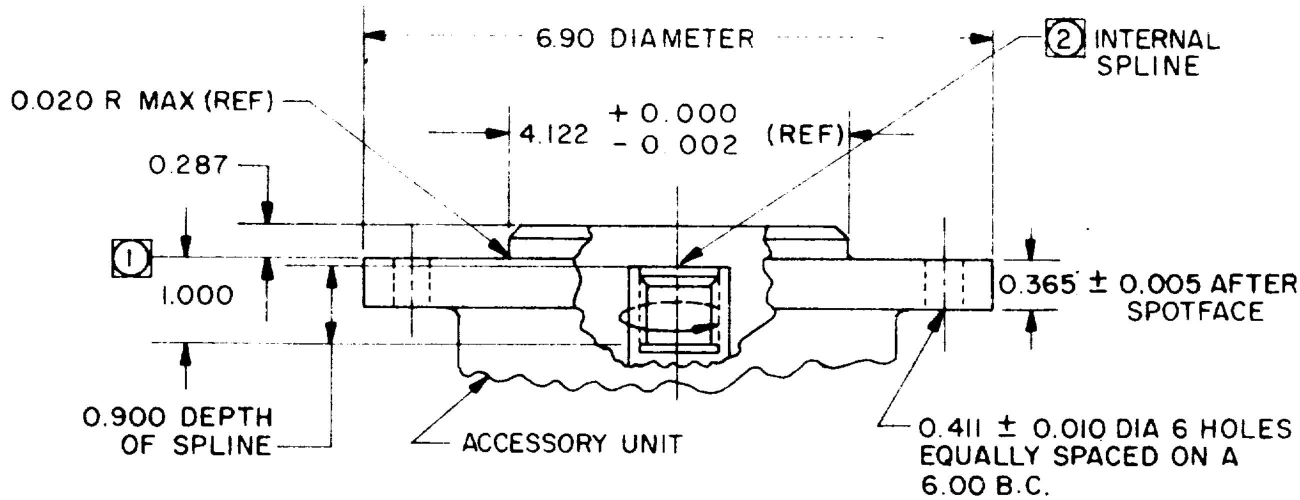
ACCESSORY DRIVE PAD

An accessory drive pad is located on the turbine exhaust manifold of the oxidizer turbopump. The pad is an adaptation of the AND 20002 standard drive pad. The basic differences are as follows:

1. An increase of the bolt circle to 6.00 in. dia and the outside to 6.90 in. dia.
2. No bearings or seals are supplied at the accessory pad.

The accessory is to be connected directly to the turbine shaft by means of a Rocketdyne provided quill. The use of this quill drive requires a female spline on the accessory. Details of the pad requirements for the accessory are shown in Fig. 10.1. The turbine manifold temperature will be approximately 610 F and the accessory will be subjected to the gas pressures within the manifold. This pressure will be approximately 30 psia and will consist of hydrogen rich steam. The rotational speed of this accessory drive is 8950 $^{+620}_{-440}$ rpm during mainstage with rotation being clockwise viewing the engine drive pad. Torque on this accessory drive shall be held at an absolute minimum at start. The point in the start sequence at which the significant power may be drawn from this pad is a subject of review with Rocketdyne.

The engine will be delivered with the accessory pad blanked off. The quill shaft will be separately packaged.



NOTES:

1. THIS SKETCH NOT TO SCALE
2. PUMP PAD PER AND 10262 TYPE XII - H EXCEPT AS SHOWN.

[1] 3. THIS SURFACE TO BE FLAT WITHIN 0.004 TIR AND TO HAVE A 32 RMS CONCENTRIC FINISH WITHIN 5.375 DIA.

[2] 4. PUMP TO BE SUPPLIED WITH AN INTERNAL SPLINE 20/30 PITCH, 30 DEG PRESS ANGLE. 16 TEETH 0.800 PD (THEOR.)

5. OIL DRAIN HOLES TO BE OMITTED.

Figure 10.1 Detail of Accessory to Mate With Engine Accessory Drive

The engine performance balance allows for 30 hp to be extracted from the pad while the engine is operating at mainstage turbine speed. Exceeding this power value will result in a reduction in other performance figures. However, the pad and drive system is structurally suitable for 100 hp extraction at mainstage speed.

The 30 hp value is based on nominal power requirements to gimbal the engine at model specification limits. If hydraulic power is selected, it is assumed that peak loads in the hydraulic system are absorbed by a properly selected vehicle system accumulator.

The following limits of drive pad capabilities must be observed:

1. Maximum starting torque 100 in. lb
2. Maximum running torque 700 in. lb
3. Maximum accessory weight 25 lb
4. Maximum overhung moments 125 in. lb

The estimated liquid oxygen pump speed buildup during the engine start transient is presented in Fig. 10.2.

HYDROGEN TANK PRESSURIZATION SYSTEM

The engine will supply hydrogen gas for tank pressurization at a rate of 3.00 lb/sec at a temperature of -260 F at the nominal engine operating level and nominal propellant utilization valve setting. The pressurant will be obtained by bleeding hydrogen from the thrust chamber cooling jacket upstream of the injector.

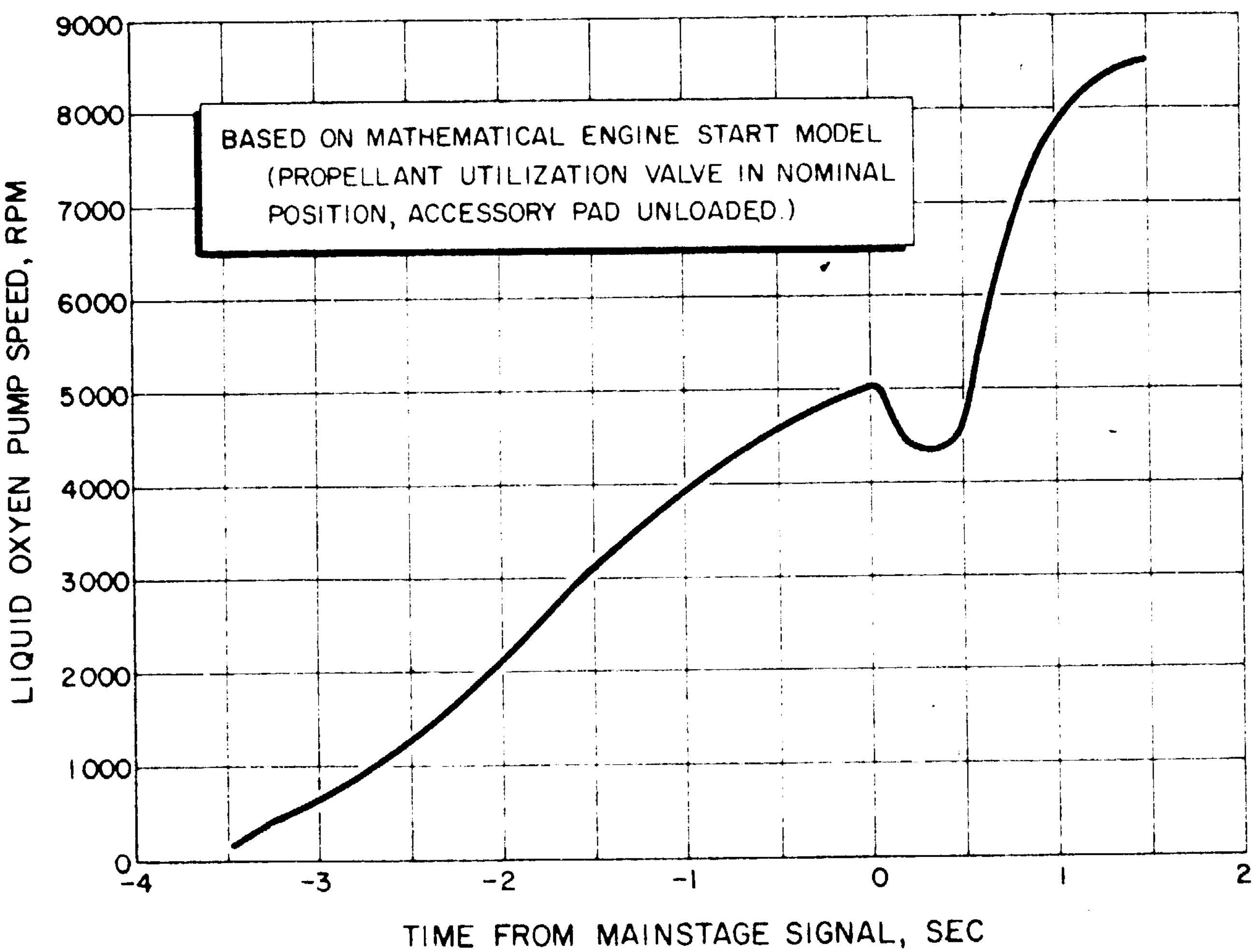


Figure 10.2. Estimated Oxygen Pump Speed During Engine Start Transient at Altitude

OXYGEN TANK PRESSURIZATION SYSTEM

The engine will supply oxygen gas for tank pressurization at a rate of 3.90 lb/sec at a temperature of -170 F at the nominal engine operating level and nominal propellant utilization valve setting. The pressurant will be obtained from an oxygen heat exchanger located in the oxidizer pump turbine exhaust gas duct.

PRESSURIZATION SYSTEM FLOWRATE VARIATION

Variation of pressurization system flowrates will alter the engine operating level. The incremental changes in engine performance due to independently decreasing the tank pressurization flowrates from their nominal values to zero are:

	Hydrogen System	Oxygen System
Engine Thrust, lb	+1500	-1300
Engine Specific Impulse, sec	+1.25	+0.55
Engine Mixture Ratio	-0.10	-0.02

The design of the oxygen pressurization system is such that a zero flowrate condition cannot be tolerated by the engine. Accordingly, a minimum flowrate of approximately 1.0 lb/sec through the heat exchanger is necessary. Figure 10.5 presents the estimated liquid oxygen heat exchanger operating line.

In computing vehicle tanking mixture ratio, the tank pressurization flow requirements should be added to the engine propellant consumption as engine mixture ratio and specific impulse do not include the pressurization flowrates.

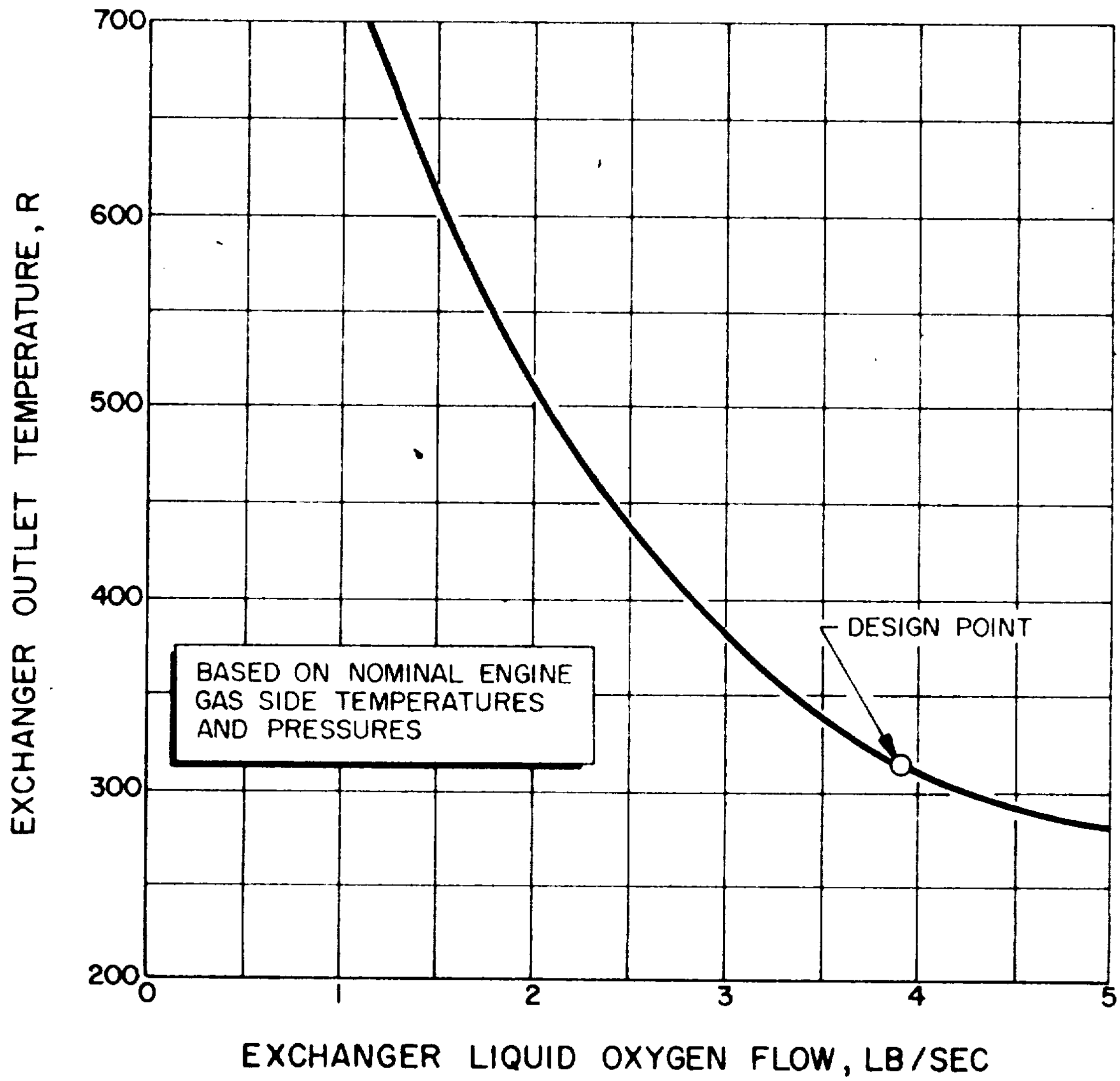


Figure 10.3. Liquid Oxygen Heat Exchanger Operating Line

CONTROLS

PROPELLANT UTILIZATION CONTROL

PROPELLANT UTILIZATION CONTROL SYSTEM

The propellant utilization (PU) control system is a device for obtaining maximum utilization of vehicle propellants by adjusting the engine mixture ratio to simultaneously exhaust the supply of both vehicle tanks. An example of its operation is presented in the following paragraph.

When the vehicle propellant level signal system indicates that the oxidizer supply is being reduced faster than scheduled, it is intended that the vehicle control system will transmit a valve opening signal to the PU system servomotor. This will cause the PU valve to open the oxidizer pump bypass line and therefore increase the rate of oxidizer circulation around the pump. This condition results in a decrease in oxidizer flow from the vehicle tank. Response time of the propellant utilization valve is 1 sec for full travel from stop to stop.

The PU valve and its servomotor are supplied with the J-2 engine. A position feedback potentiometer with 2000 ohm resistance is supplied integral with the PU valve assembly.

A mixture ratio variation limit of 10 percent above or 10 percent below nominal has been provided. Nominal flow through the valve is 65 lb/sec of oxygen. The valve has modulating capacity to vary the flowrate to $^{+55}_{-65}$ lb/sec.

Figure 11.1 shows a schematic of the electromechanical PU system. Mechanical elements and the vehicle manufacturers portion of the system are not necessarily represented by the schematic.

Electrical power requirements are as follows:

Servomotor	
Input, vac	110
Frequency, cps	400
Power required, w	100
Power source	vehicle propellant utilization controller

A propellant utilization system of the nature provided for is one method of obtaining good utilization of vehicle propellants; however, the use of this system is not mandatory with the J-2 engine should adequate utilization be obtained by engine close calibration or by other means. J-2 engine performance could be improved by deletion of the PU system.

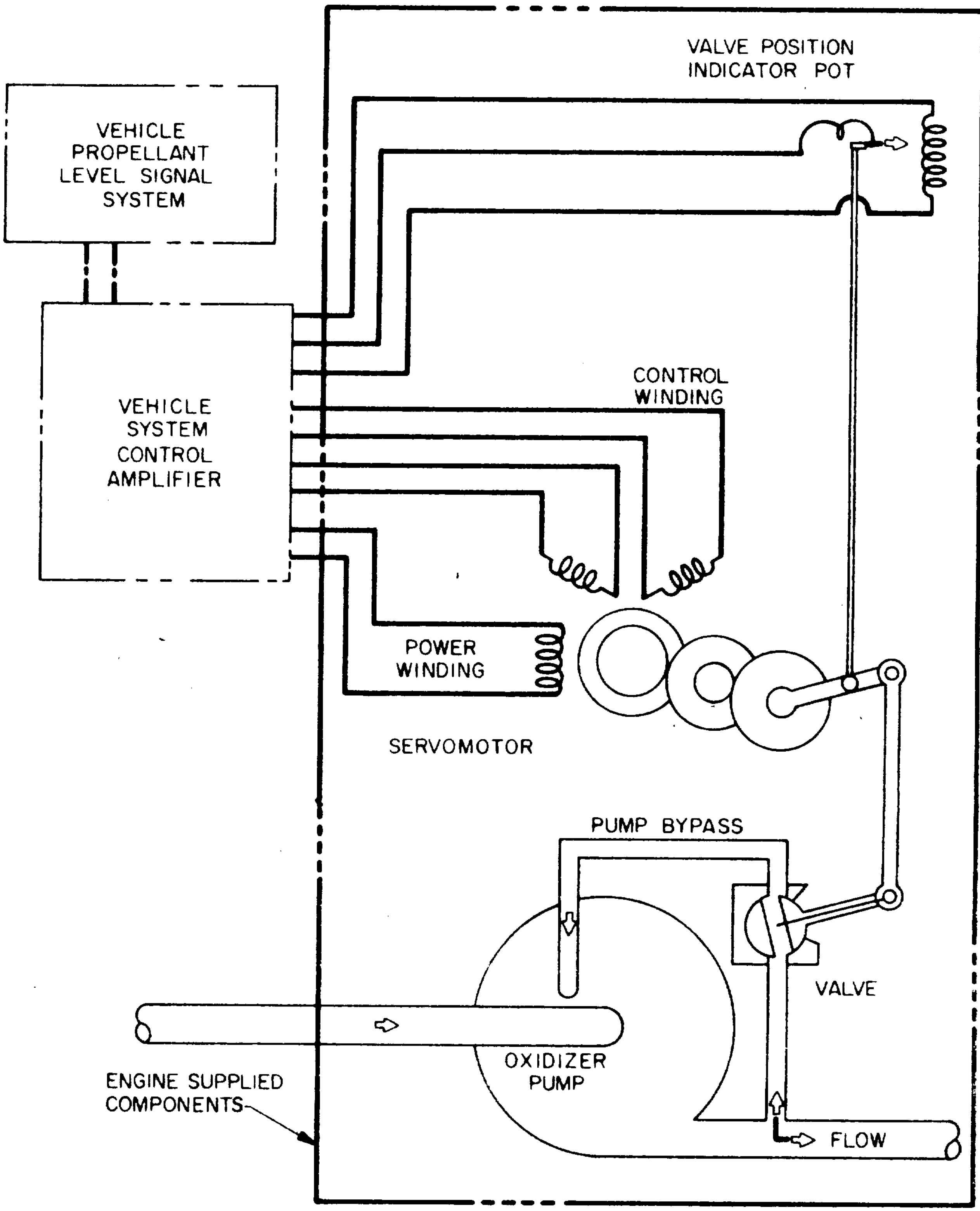


Figure 11.1 . Propellant Utilization System Schematic

HANDLING

PROPOSED AEROSPACE GROUND EQUIPMENT

Aerospace ground equipment (AGE) proposed for use with the J-2 engine is as follows:

CHECKOUT EQUIPMENT

1. Pneumatic Panels
2. Flowraters
3. Test Panel
4. Control Panel
5. Overspeed Trip Checkout Unit
6. Spark Ignition System Checkout Unit

HANDLING EQUIPMENT

1. Engine Handler
2. Handling Sling
3. Sling Adapter No. 1
4. Sling Adapter No. 2
5. Security Cover (Moisture and Dust Repellant)

6. Turbopump Inlet Support (Fuel)
7. Turbopump Inlet Support (Oxidizer)
8. Thrust Chamber Protective Pad

TEST EQUIPMENT

1. Thrust Chamber Throat Plug
2. Turbine Exhaust Plug
3. Liquid Oxygen Inlet Test Plate
4. Fuel Inlet Test Plate

ROCKET ENGINE MANUFACTURERS HANDBOOK

In accordance with the J-2 engine contract, Rocketdyne will prepare operating instructions for the purpose of directing engine maintenance, detail handling and usage. These instructions will be issued through the cognizant government agency.

STORAGE

The engine may be stored or transported at temperatures between -20 and +140 F. Reliability and engine life will not be degraded after three years of storage in this temperature range when maintained in accordance with conditions set forth in the rocket engine manufacturers handbook (for maintenance).

DOMESTIC SHIPPING

The engine will be shipped mounted to a handling fixture which is designed to mount on Air Force Transportation Trailer FSN NBAD 1450-726-1119 (Air Logistics Type 1210). Unless it is otherwise stipulated, the engine will be mounted to the trailer, shrouded for protection from the elements, and lashed to the carrier vehicle.

Dimensions of the engine envelope, mounted to the handling fixture, are as follows:

Length, in.	116
Width, in.	80
Height, in.	88

Dimensions of the engine envelope, mounted to the trailer, are as follows:

Length, in.	152
Width, in.	96
Height, in.	120

HANDLING LOADS

The engine is designed to withstand handling loads of 4 **g** in any direction.

ENGINE BUILDUP

The handling fixture used in shipping is suitable with some limitations for use as a workstand in the users facility. The limitation of this use is imposed by the horizontal engine positioning and by absence of a rotational provision.

For buildup or for storage in a vertical position, the engine may be rested on a flat pallet provided as aerospace ground equipment.

The engine is equipped with fittings for attachment of vertical and horizontal hoisting equipment ot provide flexibility in positioning the engine for buildup, loading, or installation. A cable is provided for vertical handling and a sling for horizontal handling. To transfer from horizontal to vertical positioning, two hoists are required.

Horizontal hoist fittings are located in the X coordinate axis plane on the positive side. The handling fixture mounting fittings are in the same plane on the opposite side. The engine axes are shown in Table 5.1.

HANDLING DURING INSTALLATION

The previously described horizontal position sling is the only provision made on the J-2 engine for installation in a vehicle although the engine is suitable for standing vertically on a flat and hoisting into position with a lift truck. When using the horizontal sling for multiengine installation, it is assumed the vehicle stage will be angularly located to facilitate moving the engine into position with the X plane positive side up. It is recommended that engine handling procedures be coordinated with Rocketdyne engineering representatives.

BLEEDS, VENTS, AND DRAINS

SEAL AND BLEED CONFIGURATION

DRAIN, BLEED, AND VENT PORTS

All separable hot gas and propellant flanges, and connections in the J-2 engine configuration, are equipped with a dual static seal containing an intermediate drain port to provide a leakage measurement capability (Fig. 13.1). Items 1 through 35 (Table 13.1) are of this type. The threaded 1/8 in. drain port may be routed overboard through a leakage measuring device.

The dynamic seal drain ports on both main propellant valves are equipped with vent port check valves (Table 13.1 items 36 through 39). Hydrogen leakage from the main propellant valve into an enclosed boattail (i.e., prior to stage separation) is precluded by utilization of a burst diaphragm in the valve design. Because the main oxidizer valve shaft seal vent ports are located below the main gate lipseal, liquid oxygen leakage from these vent ports can only occur through failure of the main gate seal.

To ensure the quality of propellants (all liquid) required at the gas generator for engine starting, propellant bleed valves (liquid oxygen and hydrogen) are located at the gas generator inlets. The liquid oxygen

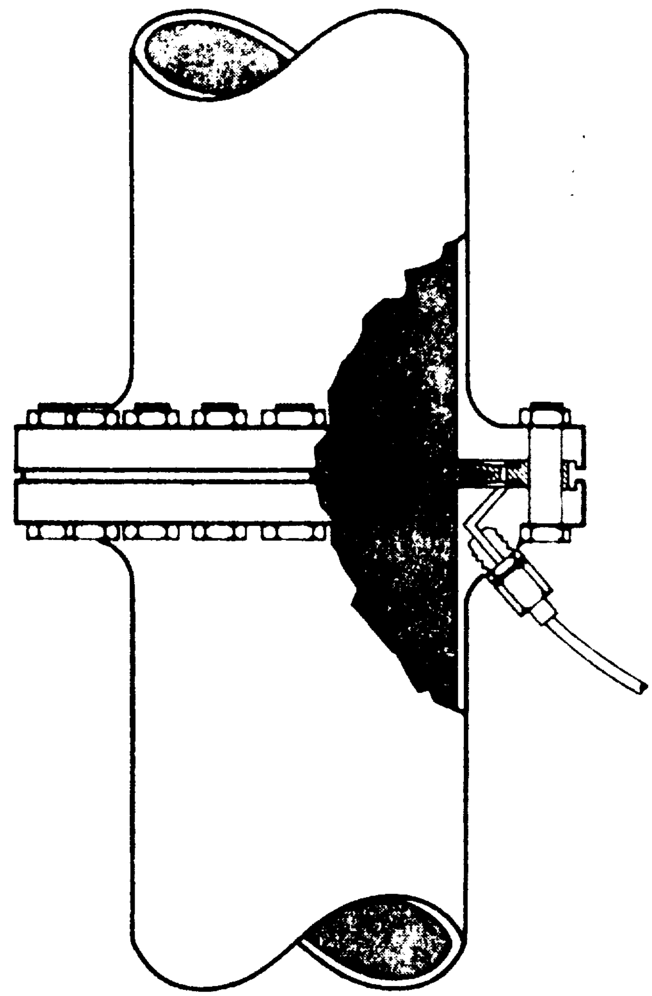


Figure 13.1. Typical Flange Seal

TABLE 13.1

J-2 ENGINE SYSTEM SEALS AND BLEED ROUTING

Item	Location	Media	Type*	Routing
1.	Fuel Pump Inlet	Hydrogen	A	To measuring device or capped
2.	Fuel Pump Outlet			
3.	Fuel Pump Volute to Rear Bearing Carrier			
4.	Fuel Pump Rear Bearing Carrier to Turbine Manifold	Hot Gas		
5.	Fuel Pump Turbine to Turbine Exhaust Hood			
6.	Fuel Turbine Torquing Pad			
7.	ASI Fuel Valve to High Pressure Duct	Hydrogen		
8.	Fuel High Pressure Duct			
9.	Fuel F/M Temperature Pickup			
10.	Fuel F/M Pressure Pickup			
11.	Fuel Valve Inlet			
12.	Fuel Valve Outlet			
13.	Fuel Pump Turbine Exhaust Hood to Interconnect Duct	Hot Gas		
14.	Interconnect Duct to Oxygen Pump Turbine Inlet			
15.	Oxygen Pump Turbine to Turbine Exhaust Hood			
16.	Oxygen Pump Turbine Exhaust Duct Thrust Chamber Manifold	Hot Gas	A	To measuring device or capped

TABLE 13.1

(Continued)

Item	Location	Media	Type*	Routing
17.	Liquid Oxygen Pump Accessory Drive Pad	Hot Gas	A	To measuring device or capped
18.	Oxygen Pump Inlet	Liquid Oxygen		
19.	Oxygen Pump Outlet			
20.	PU Valve to Oxygen Pump			
21.	Oxygen Pump Volute to Rear Bearing Carrier			
22.	Oxygen Pump Rear Bearing Carrier to Turbine Manifold	Hot Gas		
23.	Oxygen High Pressure Duct	Liquid Oxygen		
24.	Oxygen F/M Temperature Pickup			
25.	Oxygen F/M Pressure Pickup			
26.	Oxygen Valve Inlet			
27.	Oxygen Valve Outlet			
28.	Oxygen Valve to ASI Liquid Oxygen Valve			
29.	Oxygen Valve to Liquid Oxygen Dome Purge Check Valve			
30.	GG Valve to Liquid Oxygen Bleed Valve			
31.	GG Oxidizer Side to Injector			
32.	GG Valve to Fuel Bleed Valve	Hydrogen	A	To measuring device or capped

TABLE 13.1

(Continued)

Item	Location	Media	Type*	Routing
33.	GG Fuel Side to Injector	Hydrogen	A	To measuring device or capped
34.	Thrust Chamber to Dome	Hydrogen		
35.	Thrust Chamber ASI Assembly to Thrust Chamber Dome	Hot Gas		
36.	Fuel Valve Gate Shaft	Hydrogen	B	**Vent port check valve. Not plumbed
37.	Fuel Valve Actuator Housing	Hydrogen and Helium		
38.	Oxygen Valve Gate Shaft	Oxygen		
39.	Oxygen Valve Actuator Housing	Oxygen and Helium		
40.	GG Liquid Oxygen Bleed Valve Drain Line	Oxygen	--	Routed overboard through vehicle skin
41.	GG Fuel Bleed Valve Drain Line	Hydrogen	--	Routed to customer connection. (Provision must be made by the vehicle contractor to vent this line to atmosphere for engine bleeding, prior to engine start.)

TABLE 13.1

(Continued)

Item	Location	Media	Type*	Routing
42.	Liquid Oxygen Pump Seal Bleed	Oxygen	B	Routed overboard through customer connection
43.	Liquid Oxygen Pump Turbine Seal Bleed	Hot Gas	B	Routed along thrust chamber contour and vented to atmosphere
44.	Hydrogen Pump Turbine Seal Bleed	Hot Gas	B	Routed along thrust chamber contour and vented to atmosphere

*An A-type seal is a dual static seal with an intermediate vent port for leakage measurement capability

A B-type seal is a dynamic seal

**It is not necessary to plumb these drain ports overboard bleed line (Table 13.1, item 40) and the fuel bleed line (Table 13.1, item 41) are to be plumbed through a customer connection to the vehicle skin and overboard. Provision for venting these lines to atmosphere, prior to engine start, must be provided by the vehicle contractor.

Two bleed lines from the liquid oxygen turbopump (Table 13.1, items 42 and 43) must be bled to a point of identical back pressure. These bleeds drain two side-by-side cavities, one containing hot turbine gases and the other liquid oxygen. The oxidizer pump seal bleed will be plumbed overboard through a customer connection, and the liquid oxygen pump turbine seal bleed will be plumbed to atmosphere along the thrust chamber contour in the vicinity of the chamber exit.

The hydrogen turbine hot gas bleed line (Table 13.1, item 44) will be plumbed to atmosphere along the thrust chamber contour in the vicinity of the chamber exit.

CUSTOMER CONNECTIONS

CONNECT POINTS

External connect points are as follows:

1. Power input, dc
2. Power input, ac
3. Vehicle signals

   Engine Start, step function of 2.0 sec duration

   Engine Cutoff, step function of 2.0 sec duration

4. Clustering Connection

   Enables operation of a single engine or a cluster configuration

5. Instrumentation

   Provides necessary information on control system condition to ground support equipment or the vehicle

6. Helium ground fill, 4500 psi
7. Hydrogen gas to pressurize vehicle hydrogen tank
8. Oxygen gas to pressurize vehicle oxygen tank
9. Fuel inlet duct flange
10. Oxidizer inlet duct flange
11. Gimbal block
12. Gimbal actuator attach mounts

13. Fuel Turbopump seal purge

14. Fuel turbine seal purge

15. Oxygen turbine seal purge

16. Fuel turbopump seal bleed

17. Oxygen turbopump seal bleed

18. Gas generator fuel bleed valve vent

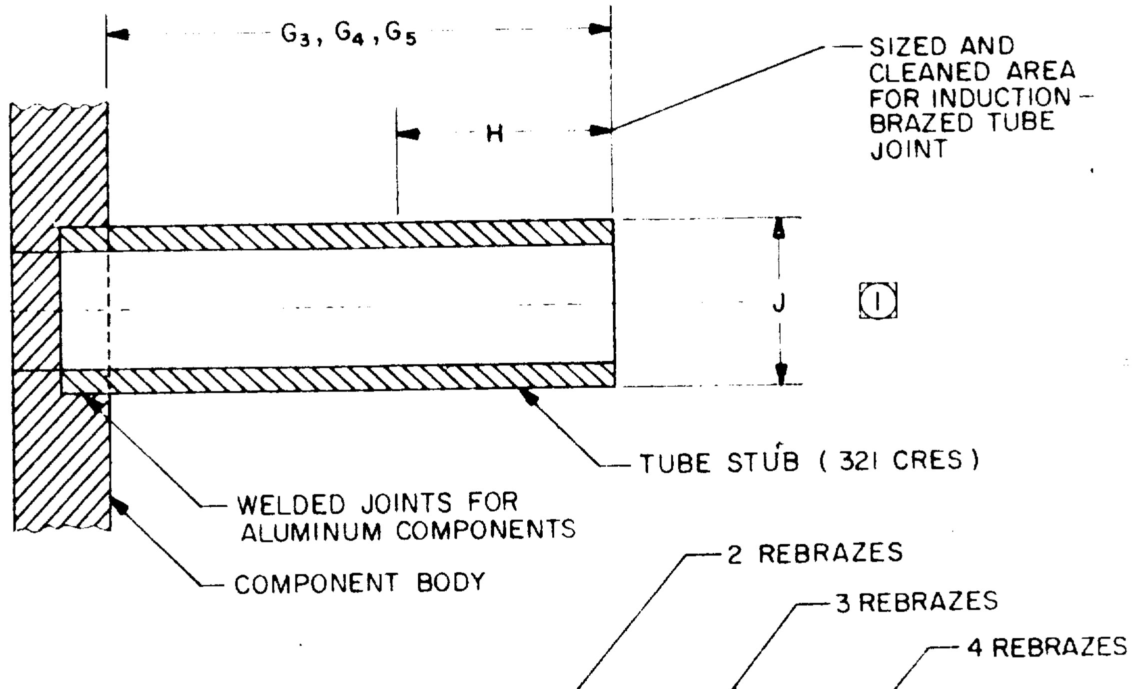
19. Gas generator oxygen bleed valve vent

To attain the degree of reliability required of the J-2 engine system, leakage from separable connections of ducts, lines, housings, etc., will be minimized through the use of brazing techniques.

Stub tubes are provided where practical for brazing engine plumbing to the vehicle to extend reliability to vehicle-to-engine connections. Allowance is made on the engine for two subsequent tube disconnecting cuts and re-brazes. If more rebrazing operations than this are required, it is expected that provisions will be made on the vehicle plumbing to permit the required number.

Figure 14.1 shows the stub lengths of tube required to make various numbers of rebrazes. Figure 14.2 shows the space requirements for a brazed joint in various tubing sizes when Rocketdyne brazing sleeves and induction heating coils are used.

These heating coil assemblies hold the tubes in position for brazing, provide for coolant gas flow through the electrodes, and provide for flow of inert gas over the joint while at elevated temperature. The coil assembly is suitable for use with a 15 kva induction heating unit.



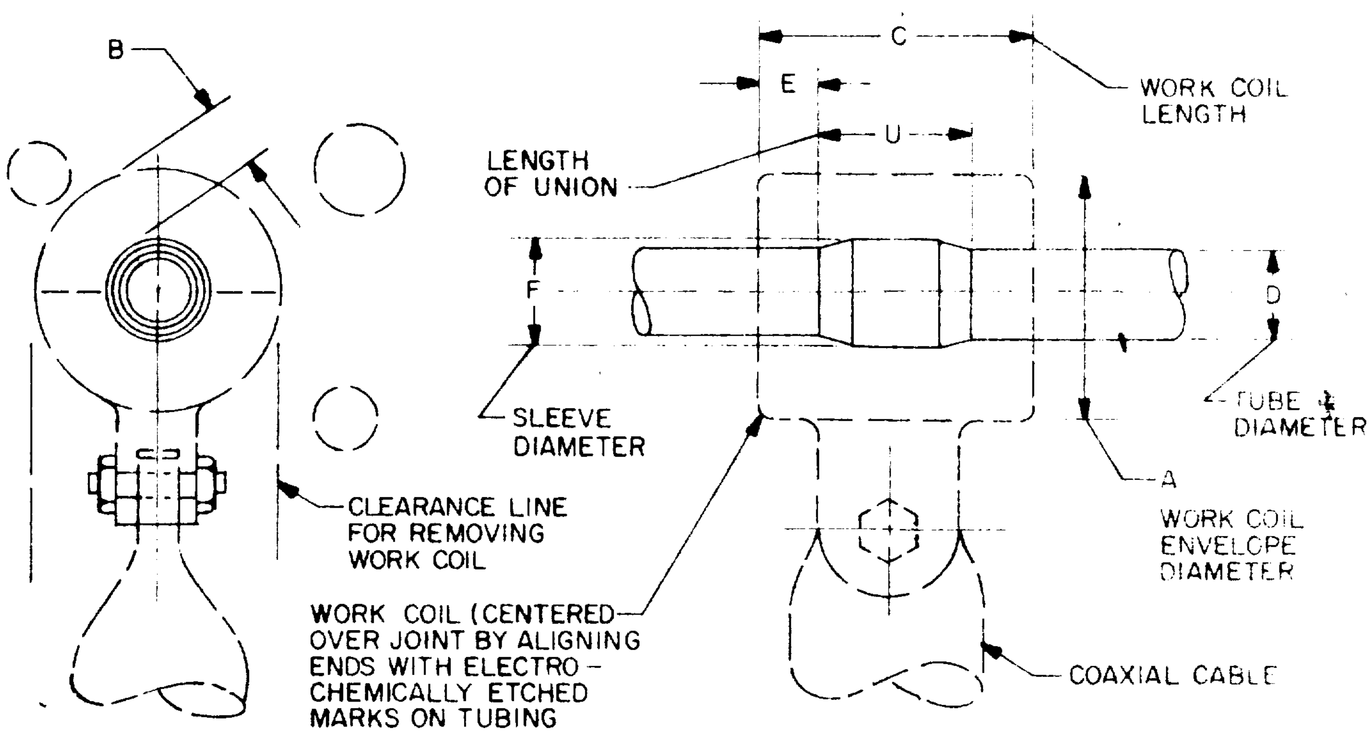
Nominal Tube Size, in.	$J^{+.003}_{-.000}$	$H^{+.060}_{-.000}$	$G_3{}^{+.060}_{-.000}$	$G_4{}^{+.060}_{-.000}$	$G_5{}^{+.060}_{-.000}$
1/8	0.135	0.437	4.219	5.344	6.469
1/4	0.260	0.468	4.328	5.484	6.640
5/16	0.322	0.500	4.440	5.628	6.816
3/8	0.385	0.531	4.548	5.767	6.986
1/2	0.510	0.625	4.877	6.190	7.503
5/8	0.635	0.687	5.094	6.469	7.844
3/4	0.760	0.781	5.923	7.517	9.111
7/8	0.885	0.906	6.849	8.690	10.631
1	1.010	1.000	7.690	9.753	11.816
1-1/8	1.135	1.125	8.127	10.315	12.503
1-1/4	1.260	1.250	8.565	10.878	13.191
1-3/8	1.385	1.375	9.002	11.440	13.878
1-1/2	1.510	1.500	9.440	12.003	14.566
		Minimum Values			

NOTES:

1. Lengths include multiple cutoff and rebraze consideration
2. H is equal to length of sleeve to allow for complete sleeve slide-on in tight locations
3. J represents the finished sized dimension (Nom + .010$^{+.003}_{-.000}$)
4. 1 Diameter to be round within 0.001 inch

Figure 14.1. Stub Tube Length

[redacted]



D	A	B	C	U	E	F $^{+.000}_{-.005}$
1/8	1.388	5/8	1.687	0.437	5/8	0.203
1/4	1.513		1.718	0.468		0.333
5/16	1.575		1.750	0.500		0.401
3/8	1.638		1.781	0.531		0.470
1/2	1.763		1.875	0.625		0.610
5/8	1.888		1.937	0.687		0.753
3/4	2.263	3/4	2.281	0.781	3/4	0.890
7/8	2.638	7/8	2.650	0.906	7/8	1.045
1	3.013	1	3.000	1.000	1	1.170
1-1/8	3.138		3.125	1.125		1.316
1-1/4	3.263		3.250	1.250		1.469
1-3/8	3.388		3.375	1.375		1.594
1-1/2	3.513		3.500	1.500		1.740

NOTE:

1. A dimensions based on swaged tube diameter (in joint area) of nominal diameter $+0.010^{+.003}_{-.000}$. Resultant tube to sleeve radial clearance is 0.0005 to 0.0025 in.
2. Dimensions A, B, C, and E represent acceptance minimum values.

Figure 14.2. Brazed Joint Space Requirements

[redacted]